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The History of Structural Fatigue Testing at Fishermans Bend Australia

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Air Vehicles Division
Defence Science and Technology Organisation

ABSTRACT

This report presents the history of fatigue research at DSTO's Fishermans Bend Australia facility from the early days in the 1940s when Mr. H.A. Wills, Head of the then Structures Division, foresaw with remarkable insight the impending danger of fatigue in aircraft structures. He presented an historic paper at the Second International Aeronautical Conference in 1949 and instituted a comprehensive programme of research on the fatigue of materials and structures which was vindicated within the next decade as fatigue failures began to plague first civil and then military aircraft fleets world wide.

DSTO is still a leading world authority on the fatigue of aircraft structures, as many of these research programmes have won international recognition and as fatigue investigations expeditiously undertaken for the RAAF (and at times Civil authorities) have supplied valuable information to the aircraft manufacturers, operators and researchers.

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Executive Summary

Today the results of full-scale fatigue testing (FSFT) of aircraft structures are generally considered to be the foundation for continuing structural airworthiness. DSTO can rightly claim to be a pioneer in this and related sciences. These investigations had a significant influence on the fundamental understanding of fatigue, the development of fatigue design philosophies, theories, testing techniques and technologies, and airworthiness criteria in the aeronautical sphere. Today fatigue plays the major role in the design of optimised flight vehicles. This report presents a brief history of fatigue research at DSTO's Fishermans Bend Australia facility from the early days in the 1940s when Mr. H.A. Wills, Head of the then Structures Division, foresaw with remarkable insight the impending danger of fatigue in aircraft structures.

FSFT of airframes commenced shortly after the establishment of the then Aeronautical Research Laboratory in 1939, prompted by the results of aircraft accident investigations. In the intervening years numerous fatigue tests have been successfully conducted at the Fishermans Bend site. These range from the relatively simple tests on Mosquito wings, through to the exhaustive tests on over two hundred Mustang wings, and on to the recently completed simultaneous application of dynamic buffet and manoeuvre loads test on the aft fuselage and empennage of a F/A-18.

Today's DSTO is proud of its contribution to the through life structural integrity support of (mainly) Australian Defence Force platforms (and to a lesser extent the civil aviation industry) through numerous full-scale fatigue programmes conducted at the "Bend". These have demonstrated that despite better design and analysis tools being available in the latter years, the highly optimised nature of modern aircraft coupled with increased operational requirements will likely lead to surprises, in the form of structural failures, that are better discovered in the test laboratory than through post-accident investigation.

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Mr. Molent graduated in 1983 with a Bachelor of Engineering (Aeronautical). Since commencing employment at the then Aeronautical Research Laboratories in 1984, Mr. Molent has worked in the fields of aircraft structural integrity, structural and fatigue testing, advanced bonded repair, aircraft accident investigation and aircraft vulnerability. He has over 100 publications in these technical areas. He has been attached to both the Civil Aviation Department (1985) and the US Navy (NAVAIR, 1990-1991) as an airworthiness engineer. Mr. Molent completed a Graduate Certificate in Management in 1997. Mr. Molent is currently a Principal Research Scientist and Head, Structural Integrity (Combat Aircraft), and manager, International Follow On Structural Test Project (IFOSTP), at the Air Vehicles Division.

Contents

Abbreviations and Notation

AOA	Angle Of Attack
ARDU	RAAF Aircraft Research and Development Unit
ARL	Aeronautical Research Laboratories or Laboratory
AMARC	Aircraft Maintenance and Regeneration Center
AMRL	Aeronautical and Maritime Research Laboratory
AVD	Air Vehicles Division
CB	Centre Barrel (F/A-18)
CBR	Centre Barrel Replacement (F/A-18)
CF	Canadian Forces
CPLT	Cold Proof Load Test
CSIR	Council for Scientific and Industrial Research
DADTA	Durability And Damage Tolerance Assessment
DCA	Department of Civil Aviation
DH	de Havilland Aircraft (UK)
DOA	Department of Aviation
DSTO	Defence Science and Technology Organisation
ESH	Equivalent Service Hours
FALSTAFF	Fighter Aircraft Loading Standard For Fatigue evaluation
FAR	Federal Aviation Regulations
FINAL	Flaw IdeNtification through the Application of Loads
FSFT	Full-Scale Fatigue Test
GAF	Government Aircraft Factories
IFOSTP	International Follow-On Structural Test Project
LEX	Leading Edge extension (F/A-18)
LIF	Lead-in Fighter
LOTEX	Life Of Type Extension
OEM	Original Equipment Manufacturer
MSD	Multiple Site Damage
NACA	National Advisory Committee on Aeronautics
NDI	Non-Destructive Inspection
NPL	National Physics Laboratory
QF	Quantitative Fractography
RAAF	Royal Australia Air Force
RAE	Royal Aircraft Establishment
R&D	Research and Development
RST	Residual Strength Test
SFH	Spectrum Flight Hours
USN	United States Navy
WA	West Australia
WOM	Wing Optimisation Modification (F-111)

1. Introduction

The Aeronautical Research Laboratory (ARL) was established at a site on Fishermans Bend Victoria Australia [1] as a division of Council for Scientific and Industrial Research (CSIR) in 1939 following the recommendation of Mr. Wimperis, formerly Director of Scientific Research for the Air Ministry in Britain. Mr. Wimperis was commissioned to advise the Australian Government on the inauguration of aeronautical research in Australia. Events, particularly in the field of fatigue have proved Mr. Wimperis' recommendations to be remarkably accurate, but it is interesting that he made no mention of a structures research centre. However, the first part of the Laboratories to take shape was a structures and materials section and one of the first buildings on the ARL site was a Structural Test Laboratory, later to be referred to locally as the "Wing Bay", with a reinforced concrete floor for reacting the test loads applied to full-scale structures (Figure 1). This section was headed by Mr. H.A. Wills, who was a leading Aeronautical Engineer with experience in aeronautics in both Australia and England. Mr. Wills was impressed from the beginning with the importance of fatigue in the high performance structures and materials being used in aircraft construction and his foresight and initiative was responsible for the fatigue research programme that has proved to be a major contribution to Australian Aeronautics.

Figure 1: Pouring of the reinforced concrete floor of the ARL Structural Test Laboratory

Early in 1942 the Laboratories investigated the fatigue behaviour of Cr-Mo steel tubing joined by various processes such as gas, electric arc or flash welding in connection with the local manufacture of Bristol Beaufort aircraft and the Australian Wirraway trainer, the fuselage of which were of welded tubular steel construction. Full-scale fatigue testing (FSFT) of airframes commenced shortly thereafter, prompted by the results of aircraft accident investigations. In the intervening years numerous fatigue tests have been successfully conducted at the Fishermans Bend site. Whilst ARL has experienced several changes of name (Aeronautical Research Laboratory, Aeronautical and Maritime Research Laboratory (AMRL) and currently the Platforms Sciences Laboratory), the Wing Bay is still

standing and FSFTs were conducted within it until 2003. The Wing Bay was complemented by the new Structural Test Laboratory in 1989, and was replaced in 2004 by the extension of the H.A. Wills Structure and Materials Test Centre building ("Building 2").

This report provides a brief description of the full-scale airframe fatigue tests conducted at the Bend and related technology developments. It borrows heavily from several texts, most notably those of Dr A.O. Payne [2][3]. Whilst it can be argued that the science of fatigue has progressed in the years since Mr Wimperis, the value of fatigue testing is of ever growing importance given the operational demands of modern aircraft and their highly weight/stress-optimised structures.

2. Period 1940-1950

2.1 Mosquito wing test

Following the failure of a locally made Mosquito wing on its test flight, static strength tests (Figure 2) showed that the accident was due to defective gluing but the test facility developed by Mr. W.W. Johnstone, the leader of Experimental Research in Structures Division at that time, was to prove a new and important development. An interconnected lever system was used to apply a distribution of concentrated loads to the wings loaded by high precision hydraulic jacks. This meant that the distributed load over the whole structure was applied by a central hydraulic pressure supply. Load was applied by manual operation of a hydraulic valve and the pressure supply was from the pump of a small 6,000 lb Amsler testing machine in the laboratory. However, the essential features of this system constituted a major step forward and led to the later development of a completely automatic hydraulic repeated loading system, capable of rapidly applying and controlling any desired range of load. In those early days, pumps and hydraulic valves removed from World War 2 surplus aircraft made available for test purposes were used in the construction of this equipment. Repeated load tests were then carried out on the Mosquito wings, but these tests indicated that the static strength was not appreciably affected by repeated loading [6]. However, there was evidence to indicate that the fatigue performance of metal structures was not nearly so satisfactory; in 1945 a Stinson aircraft on route from Melbourne to Broken Hill lost a wing in gusty weather and all 10 occupants were killed [5]. The subsequent investigation carried out by ARL at the request of the Department of Civil Aviation (DCA), revealed a classic fatigue failure in a welded joint in the wing spar. This showed the relevance of the research work on welded steel construction and was an early indication of the remarkable foresight of Mr. H.A. Wills in predicting the dangers of fatigue in aircraft structures.

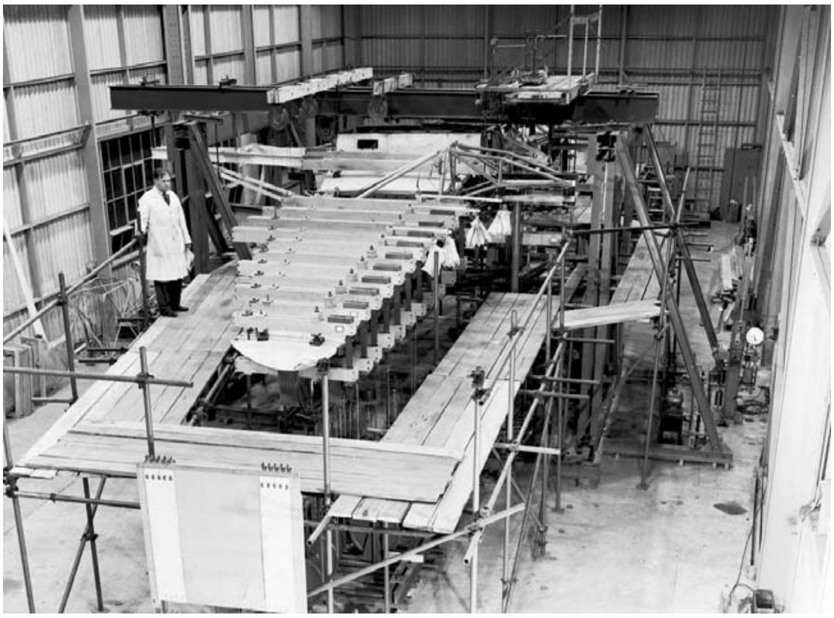

Figure 2: The hydraulic Mosquito wing test

2.2 Boomerang wing tests

The Stinson accident stimulated a research project into the fatigue behaviour of full-scale wing structures and 14 wings from the Australian Boomerang CA-12 fighter aircraft were made available by the RAAF to the Laboratories for testing [7].

An improved version of the hydraulic loading rig (Figure 3) was used to carry out these tests up to a life of 50,000 cycles, while wings were tested to much longer lives in a resonant vibration testing rig developed for this purpose. In this rig the structure was vibrated near its natural frequency, thus giving a much more rapid rate of testing (at least 10 times that of a hydraulic testing rig) with the expenditure of very little power. The technique was further developed in subsequent test series in which both masses and stiff springs were attached to the structure (see Figure 7) to modify the vibration mode so that the loading corresponded to the actual flight loading over the entire wing. At this stage the ARL fatigue testing installations and techniques were considerably in advance of the then-current practice overseas and they played a major part in the success of the fatigue work.

The investigation on Boomerang wings was completed in 1949 and it produced for the first time a complete stress-life (S-N) diagram for a full-scale fabricated structure (Figure 4). This was a remarkable achievement since systematic fatigue tests on such large structures had not been thought feasible. The results of these tests gave an insight into the complex fatigue behaviour of a complete structure and revealed substantial differences in the fatigue characteristics of structures and typical structural joints. The results also showed that the fatigue performance of complete structures was relatively low: much lower that the fatigue strength of typical structural joint specimens.

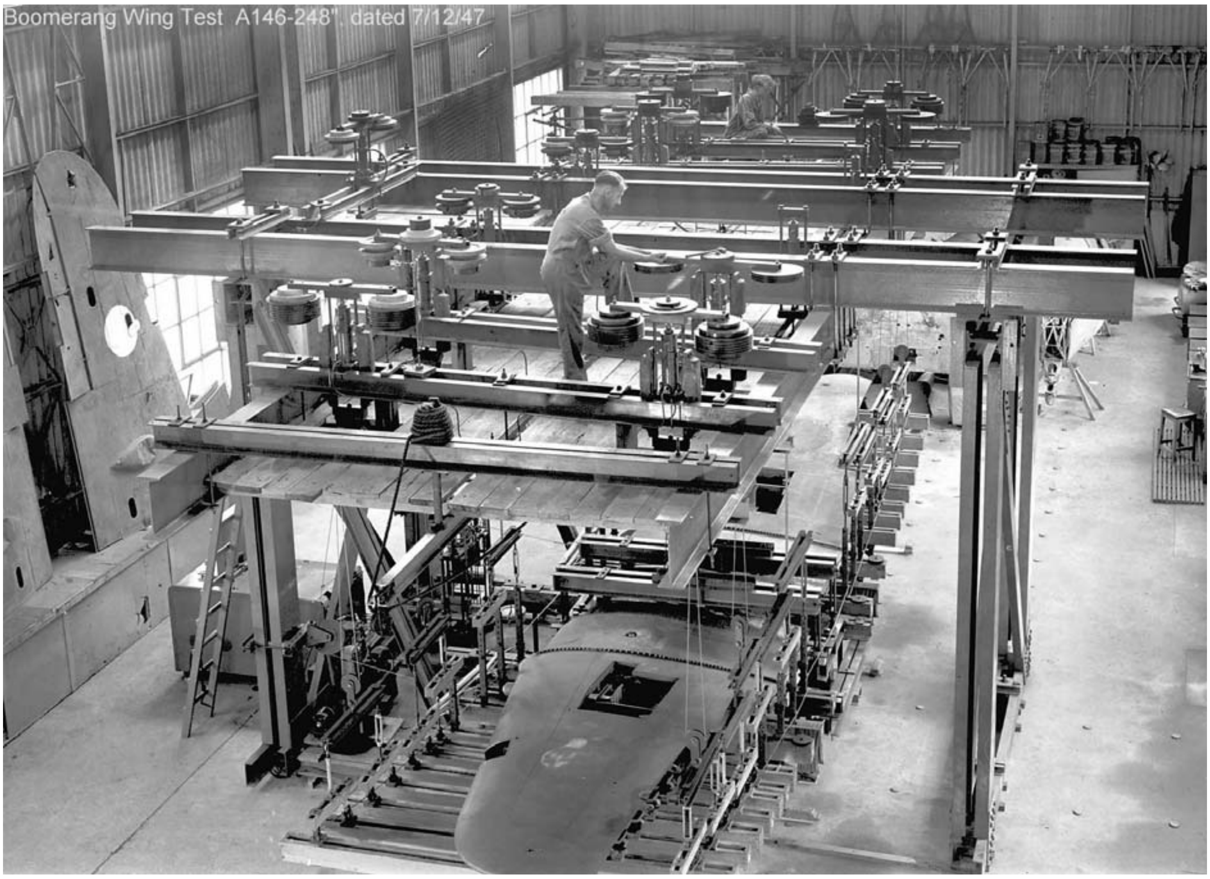

Figure 3: Hydraulic Boomerang wing test rig

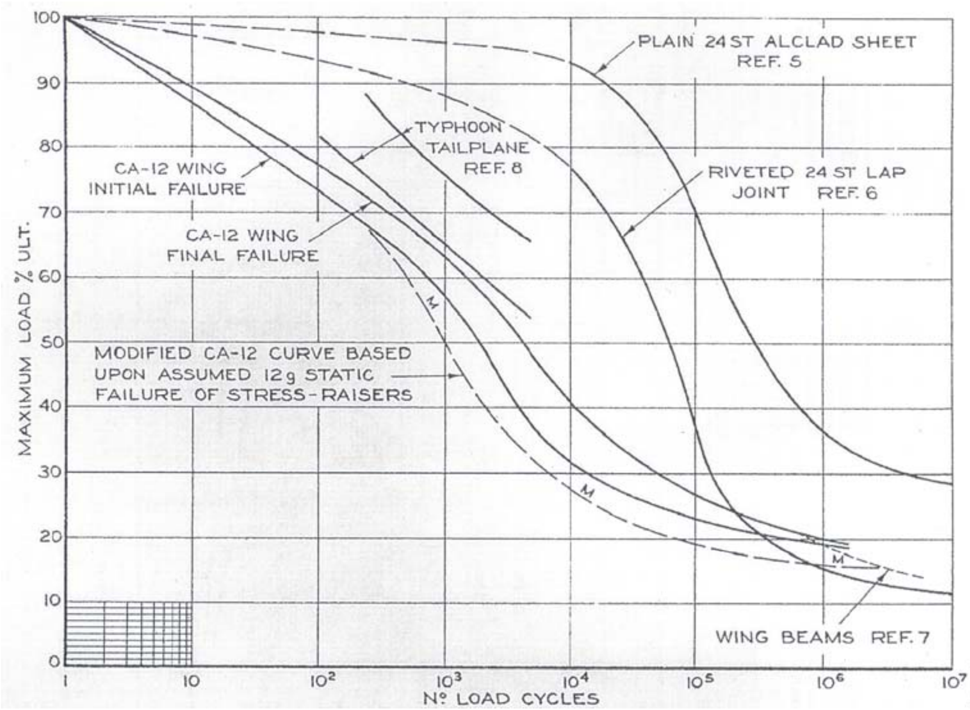

Figure 4: Comparison of CA-12 Wings and other tests (figure and REF therein from [7])

Concurrent with these investigations other work in connection with structural fatigue was underway. Dr. Hooke undertook a programme of research on the loading environment of both civil and military aircraft which provided information on the typical load history of aircraft structures. With the cooperation of the DCA, the Australian civil airlines and the RAAF, much urgently required data began to accumulate. In 1949 Mr. H.A. Wills presented a paper at the Second International Aeronautical Conference in New York [8], which proved to be a major contribution in this field. He predicted the impending danger of fatigue and listed the major contributing factors, all of which have proved to be remarkably correct even to the present day. The paper presented a comprehensive survey of the field and outlined a procedure for estimating the life of aircraft structures, which forms the basis of the methods in use today. As a result of the findings from the Boomerang wing tests, and the impact of the H.A. Wills paper, it was decided to extend the investigation on full-scale structures.

2.3 Initial Mustang wing tests

Wings from 20 Mustang P51D wings declared surplus after World War 2 were obtained and tested [2], using improved versions of both the hydraulic loading rig and the vibration loading rig (Figure 5) to give further information on the behaviour of fabricated aluminium alloy structures. However, the bulk of the Mustang wing test program was conducted in the next decade.

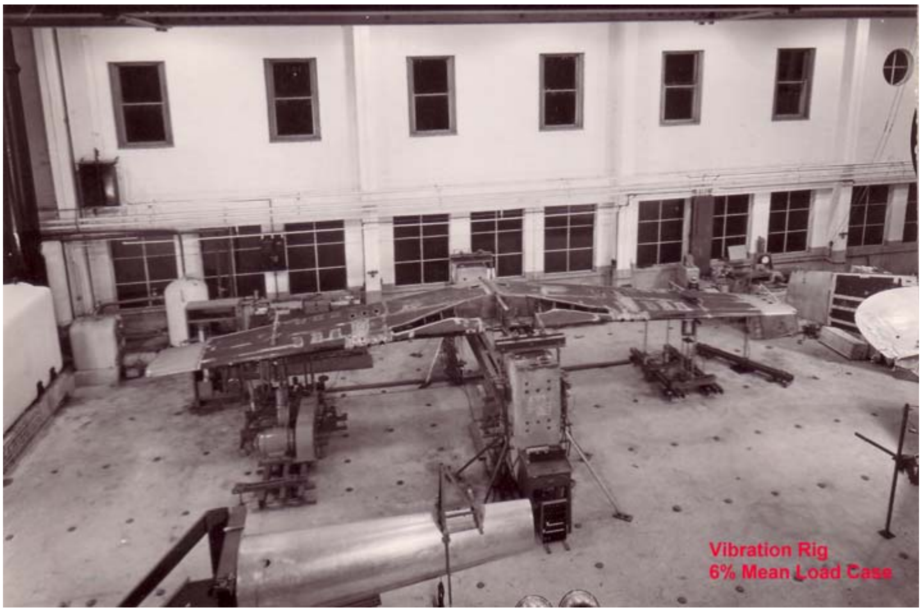

Figure 5: Early version of the Mustang wing vibration loading rig

In February 1949 ARL was transferred from CSIR to the Department of Supply. Dr M.W. Woods stated [9] "I think we liked belonging to CSIR, we enjoyed our association with scientists in other areas of the Organisation, and most of us viewed with alarm the prospect of becoming Public Servants".

3. Period 1950-1960:

While the initial Mustang tests were in progress a second catastrophic accident occurred in Western Australia (WA) and gave early evidence of the truth of Mr. Wills's predictions and greatly influenced the fatigue research programme. In October 1951 a de Havilland (DH) Dove aircraft flying from Perth to Kalgoorlie crashed 5 minutes before it was due to reach its destination, with the loss of all on board. There was no difficulty in determining the cause of this disaster; a fracture in the spar of the wing centre section. ARL was consulted on this problem by DCA and flight loads data from the W.A. routes were used with fatigue data from ARL tests to provide estimates indicating a fatigue life of between 3,000 and 12,000 hours. This was in reasonable agreement with the service failure at 9,000 hours considering the assumptions involved. However, the investigation revealed the need for more extensive loads data on Australian routes and for more comprehensive fatigue data. In fact, the only FSFT data existing at that time were the results from the ARL fatigue tests on Boomerang wings and some UK data from fatigue tests on tailplanes at the Royal Aircraft Establishment (RAE).

These developments had a major impact on fatigue research which was actively undertaken in three major areas – basic research on the fatigue of materials under Mr. J. B. Dance, the then Head of Materials Division; flight load and aircraft response research under Dr. F. H. Hooke; and structural fatigue research involving a team in the Structures Experiment Group under Mr. W.W. Johnstone. This was also a period of great stimulus

and activity in structural airworthiness in Australia and ARL and DCA produced proposals for fatigue life substantiation of aircraft structures which had an important influence on developments overseas. Furthermore the ARL-DCA collaboration was to have a profound influence on the development, in Australia, of structural airworthiness standards from the fatigue viewpoint not only in the civil field but also in military aircraft, as fatigue failures began to occur in the large military fleets.

A leading figure in the early research on fatigue in the Materials Division at ARL was Dr. A.K. Head who studied the basic characteristics of the growth of fatigue cracks and much of the later work in this area is based on this approach. He also investigated the effect of cyclic stressing on the microstructure and mechanical properties of metals. These later studies owed much to the presence in Australia of Dr. W.A. Wood who acted as a consultant to ARL following his pioneering research work in this field at NPL England. This work provided much valuable information and significantly affected the course of background research on the metal physics aspects of fatigue throughout the world. Subsequently Dr. I.J. Polmear became one of the leading research workers in the fatigue of materials and from his studies on the basic characteristics of fatigue he was led to propose the addition of small traces of elements such as silver to the aluminium alloys to improve their resistance to fatigue and corrosion.

Research work in the flight loads area led by Dr. Hooke was carried out with the co-operation of DCA and the Australian civil airlines who agreed to have the necessary recording instruments, namely the RAE counting accelerometer and the RAE fatigue meter, installed in their passenger and freighter aircraft [13][14]. This gave information on the occurrence of atmospheric turbulence on the major Australian routes and, in conjunction with similar data from overseas, enabled fatigue life estimates to be made of the later high speed civil aircraft such as the Viscount which were coming into service in Australia. The basis of the approach development by Mr. Johnstone and his team (which included A.O. Payne and J.Y. Mann [15]) was to investigate the problem experimentally by studying the fatigue behaviour of complete structures in the laboratory. The major influencing factors could then be identified and studied, while at the same time the problem yielded relevant data and provided an understanding of the fatigue behaviour of a complete structure which enabled service problems to be solved.

3.1 Mustang wing tests

The Mustang wing programme was extended with tests planned on 90 main planes obtained through the cooperation of the RAAF. Both hydraulic and vibration loading rigs were used.
Hydraulic jacks were connected to a number of loading stations on the wing through a system of levers (Figure 6). The required load distribution was thus obtained, the load being uniquely determined by the hydraulic pressure.

For tests of long duration the vibration testing technique was developed in which the complete wing was vibrated in a primary bending mode appropriate to the span wise load distribution to be applied. The structure had in general to be modified to obtain the required bending mode and in the early development of the method this was done by

adding masses. In addition the nodal points of the vibration were forced by fixing the wing at the fuselage connections, thus introducing the required alternating reactions at these points.

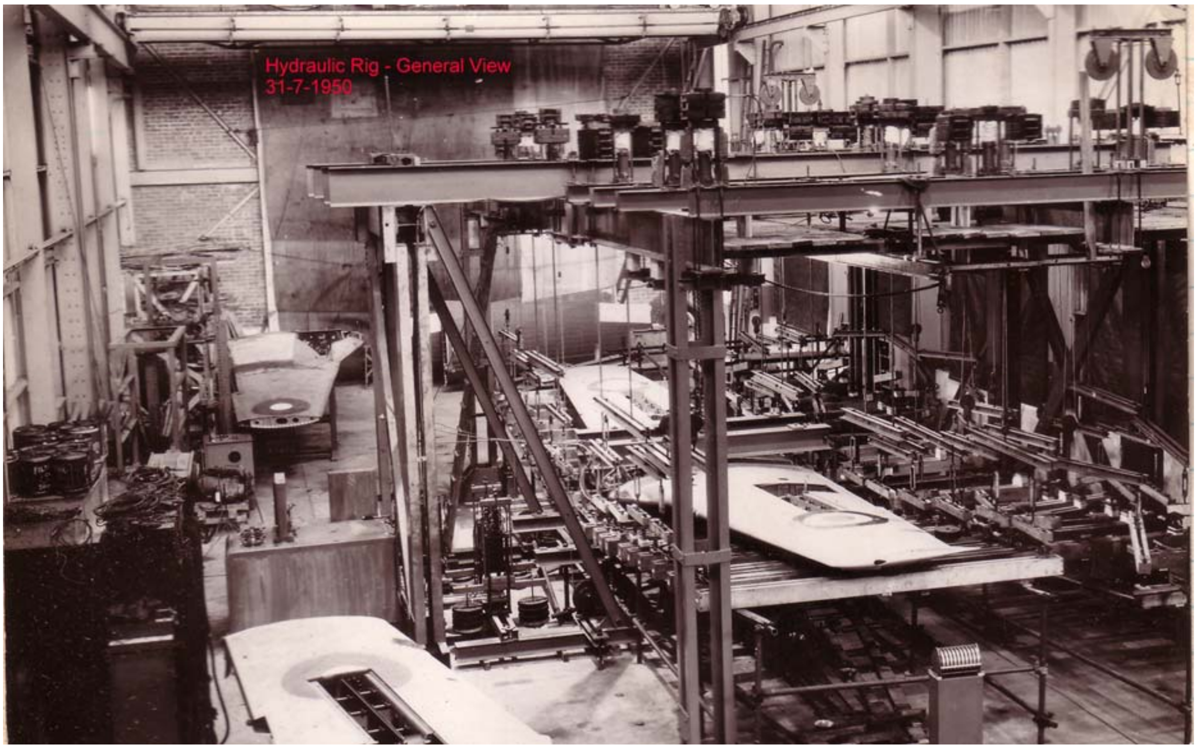

Figure 6: Hydraulic loading rig for testing Mustang wings

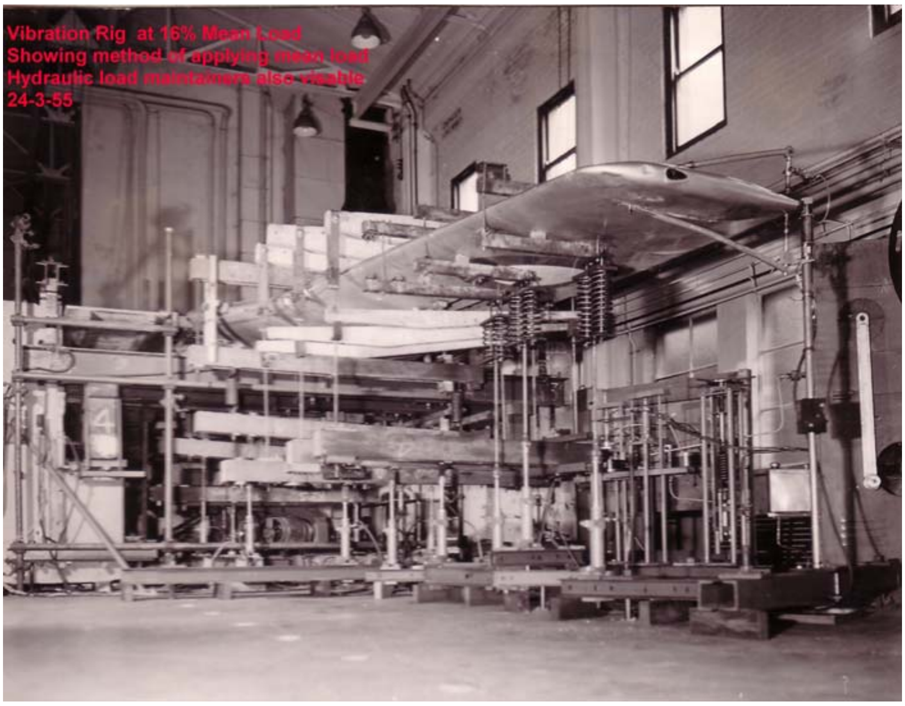

Figure 7: General view of Mustang wing vibration loading rig

This method produced the required shear distribution throughout the span, but it involved the attachment of very heavy masses, particularly near the centre-section where the deflections were small. In a further development of the technique (Figure 7) both masses and stiff springs were attached to the structure thus enabling both positive and negative shear forces to be introduced. Moreover the flexibility of the main supporting beams was designed to allow a predetermined bodily oscillation of the wing under vibration thus modifying the deflections. These two developments greatly improved the method and made the attachment of very large masses unnecessary. In a further modification of the rig designed to enable mean loads to be applied, hydraulic jacks were employed to apply the mean load to the structure through the stiff springs used to modify the vibration mode. A mechanical stroking machine driving the wing through a pair of stiff springs excited the vibration. The load was shown by a deflection indicator at the wing-tip. The technique was successfully used at relatively high load ranges, and it was thought most suited to tests at low load amplitudes for which the testing time by other means would be prohibitively long.

In an investigation of this magnitude the use of wings of varying service life was unavoidable and tests were designed to investigate the possible effects of this factor.

A series of reports were published in the mid-1950s presenting information on the fatigue characteristics of a full-scale structure [19] to [27]. A number of important results were presented on the nature of structural fatigue failure, fatigue crack propagation and the effect of high loads on fatigue performance. One of the most important and far reaching results was the discovery that there were various fatigue critical areas in a complex structure, but the one where the failure predominated depended on the load range in the fatigue test. Fatigue at that time was becoming a matter of concern in the US military fleets and structural fatigue research in that country was being undertaken particularly at NACA who obtained permission to publish some ARL reports referred to above in their own report series thus giving the ARL work wider circulation within the USA [10][11]. The Mustang programme continued to completion with tests on a total of 110 wings which, with each half tested to destruction, gave 220 test points.

The frequency distribution of the logarithm of Mustang wing fatigue life to final failure was compared with the normal distribution, which was found to be "a good approximation which is if anything conservative" [3]. The test results for each load range were standardized by subtracting the mean and dividing by the standard deviation. For Mustang wings the standard deviation has been obtained for each group of specimens tested at a particular load range. These results show considerable random variation ranging, from 0.021 to 0.308, the mean value being s = 0.12 [3]. It is thought that this variability was in part due to the usage of the wings tested. Some wings were retired from service with considerable war-time usage whilst some were still new in their shipping container. In addition, two separate testing rigs (hydraulic and vibration) were used and thus it was difficult to directly compare applied loads. The loads were compared through the measurements of strains in each structure. The difficultly here was finding a structural element that gave repetitive strain readings, since load transfer in the fabricated structures gave rise to variations even when dead weight loadings were used for calibrations. It was

also difficult to maintain load levels in the vibration rigs, and it is now believed that this was probably due to the presence of cracks that altered the stiffness of the wing/s and thus their natural frequency.

The RAAF's preferred airworthiness structural design guide is the UK Defence Standard (DEFSTAN) 00-970 [16]. This has traditionally been used to factor the results of an airframe fatigue test to provide an operational safe-life for an aircraft fleet. The standard deviation σ or s = 0.11 is used to represent the effects of aircraft construction and material quality on fatigue variability. The origin of the standard deviation is lost in time [17] [18] and cannot be re-derived. However its origins are generally thought to be the repeated fatigue tests of 1940's generation thin-skinned aircraft and representative structural joints, of which the DSTO tests on Mustang wings were a leading contributor [3]. This is supported by the closeness of the value s = 0.12 reported above. Further evidence can be deduced from the lifing of the Mirage aircraft [39] where Dr Payne stated "The Mirage wing test had an asymmetric spectrum; for aluminium alloy structures under such spectra the pooled value of *s* (sic) is 0.089". This was derived from approximately 124 test results which could be nothing other than the Mustang test data pooled to account for some of the known differences noted. Further, Dr Payne recommended that this value of standard deviation should be increased to the upper 99% confidence level, <u>viz. 0.11</u>, for reasons of conservatism, because of his uncertainty that the pooled variance for all aluminium alloy structures under asymmetric spectra adequately represented Mirage wings.

Typical crack propagation curves were also generated during the Mustang tests (see Figure 8 for example) and these indicated three characteristic stages of failure in this type of structure:

(i) The first stage leading to the propagation of a visible crack, designated as "initial failure". This constituted a significant portion of the total life and was found to be independent of the load range.

(ii) A period of crack propagation that occupied about a third of the total life; and

(iii) The final stage in which the rate of failure progressively increased until static failure occurs.

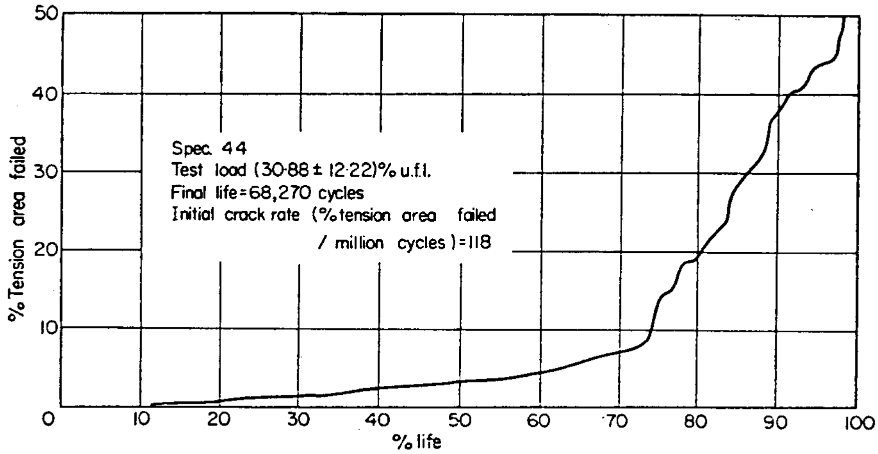

Figure 8: Crack propagation for typical gun-bay failure in a Mustang wing (from [3])

It is also of interest to note that the first application of bonded composite repair occurred on the Mustang wing tests (Figure 9) in order to repair/halt or delay crack growth [22].

At about this time (in 1953) fatigue failures occurred in the fuselage of the DH Comet which led to the loss of two aircraft with all on board. The subsequent classic investigation of the problem at the RAE led to the new technique in fatigue testing by applying repeated pressure cycles to the fuselage while filled with water and totally immersed in a specially constructed water tank. These accidents highlighted the fatigue problem and revealed to the civil aircraft industry in particular, that the predictions made by Mr. H. A. Wills some years earlier were being proved alarmingly true.

Reliable fatigue life estimates now became a major problem for DCA and ARL and this led to an extension of the Mustang fatigue investigation to include tests under complex load sequences characteristic of service conditions. These tests were done in the hydraulic loading rig, automated to run continuously day and night and controlled by a load selection device built up from telephone uniselectors which generated a random load sequence of infinite length. This enabled the fatigue behaviour of the aircraft structure under flight loading to be studied in the laboratory. It also gave for the first time, quantitative answers on the accuracy of fatigue life prediction since the basic fatigue data for the Mustang wing structure had already been obtained in the earlier investigation. In this respect the project was unique and has never been duplicated. Parallel tests were carried out on chains of notched specimens (see Figure 10) loading in synchronism with the wing to give a direct comparison with the structure. These chains were also used in other projects (see for example Figure 13). Perhaps the most important and far reaching discovery reported by ARL was that the fatigue behaviour of a complete structure under complex loading conditions could not be adequately predicted from basic fatigue data on the component material or indeed from basic data on the structure itself with any high degree of accuracy using the then current life estimation methods. This was in accordance with the earlier ARL findings on the fatigue behaviour of a structure under repeated loading and it led to the ARL recommendation for a FSFT for both safe-life and fail-safe design structures (damage tolerance design was yet unknown). This practice was by no

means generally accepted at that time and although there was some reluctance initially, it later became widely adopted in many countries.

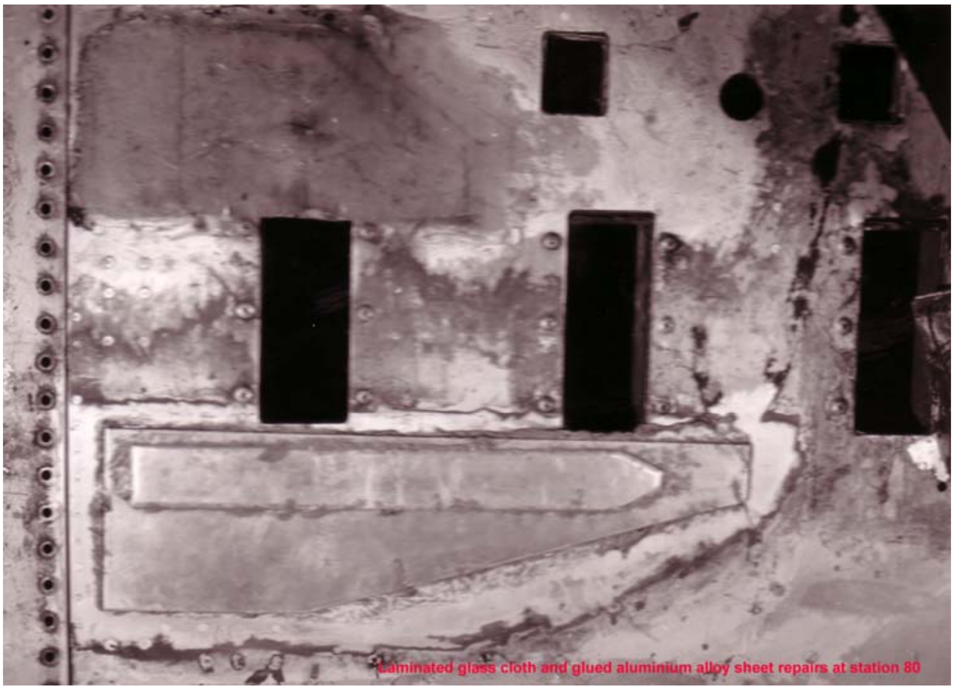

Figure 9: Early DSTO bonded repair on Mustang wing test (laminated glass cloth (above) and bonded metal sheet (below))

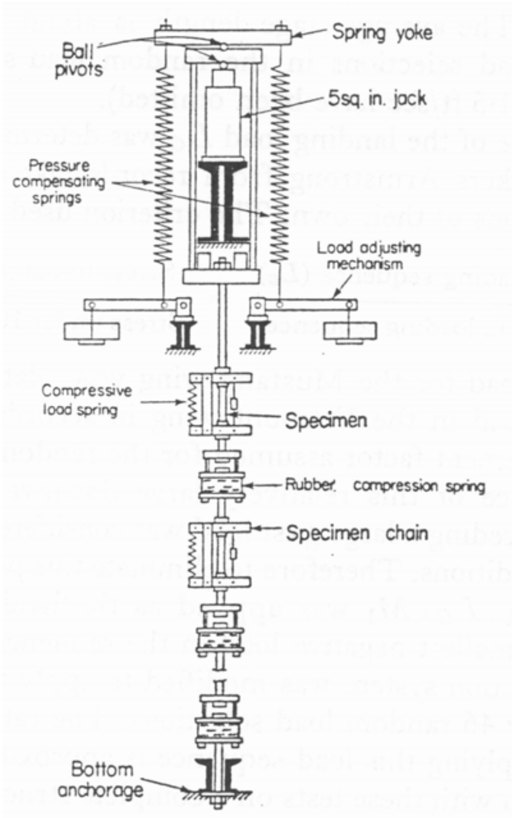

Figure 10: Diagrammatic arrangement of specimen testing rig for cumulative damage investigations (from [3])

3.2 Dove wing test

The vibration rig that had been used to carry out simple programme load tests was a development of that invented by Dr. Gassner in Germany in 1933 to represent the spectrum of service loadings. This type of test was applied at ARL to the wing of the Dove aircraft at the request of DCA to enable a mean life of over 100,000 hours to be established [28]. The DH 104 Dove aircraft had proved very successful on the regional Australian routes where they had been in operation for many years. ARL was therefore requested by the DCA in consultation with the De Havilland Aircraft Company UK to undertake a FSFT on the mainplane of this aircraft to establish the fatigue life under Australian conditions and to investigate the fatigue strength of modifications carried out during the development of the type. For the design of the test, gust load data were available from the Dove aircraft operating in WA and this was used in conjunction with S-N data obtained from the RAE as well as ground-air-ground cycles. A general view of the testing rig with the wing in position is shown in Figure 11. The testing rig was essentially the same as that employed for the fatigue tests on Mustang wings. The structure was modified to obtain the required mode of vibration by attaching masses and stiff springs at suitable rib stations. The mean load was applied through these springs by hydraulic jacks and this loading system was used to carry out static calibration of the electric resistance strain gauges attached to the structure to monitor the vibration induced loads throughout the test.

The concentrated shear and inertia loads due to engine and undercarriage inertia, being of opposite sign to the resultant air load, were applied through springs attached to the overhead structure. The frequency of vibration was approximately 8 cycles per second.

For some time the result of the ARL Dove test was used to monitor the safe-life of the aircraft throughout the world, despite DH testing carried out on the Dove in the UK. De Havilland's first test was a vibration test on an outer wing which did not contain the fatigue critical centre spar, whilst their second test was a wing test but conducted at a single load level.

An innovative testing machine was also developed by Dr. Head and Dr. Hooke to apply a complex load sequence to a small notched specimen [29]. The machine utilized the so-called white noise produced in a thermionic valve by the Brownian motion of the electrons. This signal was amplified and fed to an electromagnetic vibrator which applied corresponding load fluctuations to the specimen. The fatigue research on simple specimens had also produced important results concerning the effect of surface finish, frequency effect and the effect of pickling and anodising processes on fatigue strength. The results of the ARL work were now gaining wide recognition overseas: the vibration loading rig capable of testing a complete structure to a very large number of load cycles, the hydraulic rig equipped with a random loading device for testing wings and specimens in synchronism under representative flight conditions, and the random noise machine for testing specimens under complex load sequences were original advances which had not yet been duplicated by any other establishment: the ARL research team in testing over 220 wings had gained an extensive knowledge of the fatigue behaviour of a complete structure under various loading conditions: the basic research work on the metal physics aspects of fatigue by Dr. Wood and Dr. Head and their co-workers was also attracting considerable interest overseas: papers on the structural and material fatigue investigations had been presented by invitation at international conferences in Europe, the UK and USA: the data from the Mustang wing fatigue tests were published in the Royal Aero Society Fatigue Data Sheets.

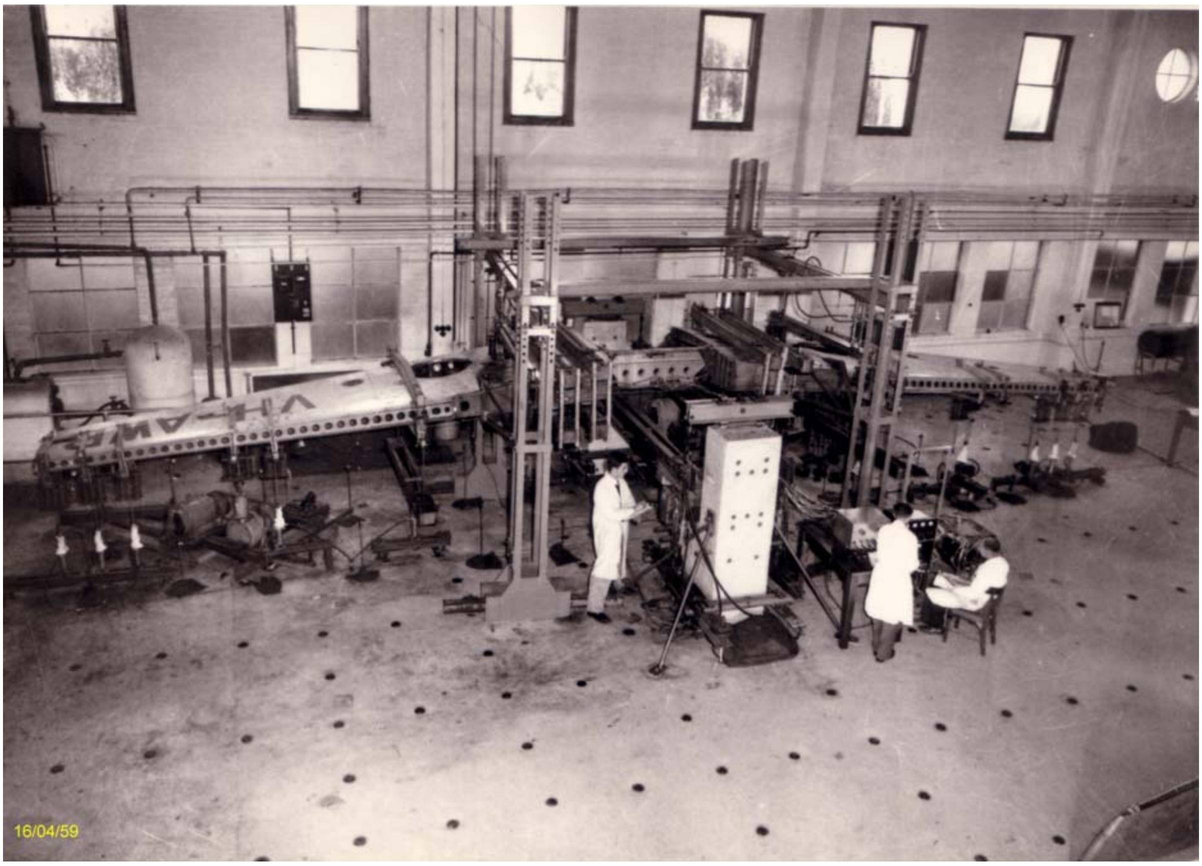

Figure 11: General view of the Dove vibration rig with wing in position for programmed load test

4. Period 1960-1975:

In the early 1960's fatigue became a major problem in the military fleets of all countries, particularly countries such as Australia which relied on a long life of type for their military aircraft. The Department of Defence took over responsibility for the laboratory in 1974 under the auspices of DSTO.

During this era major applied investigations on the General Dynamics F-111 and the Aermacchi MB326H trainer aircraft were conducted, where the properties of the new high strength materials of construction was a vital consideration, and modern equipment such as the electron microscope and the development of new inspection techniques were to play an important part in interpreting the results of fatigue tests and establishing inspection procedures.

The commitments for the RAAF became increasingly heavy as the requirement for optimum fatigue life necessitated major investigations of the fatigue performance of aircraft structures and materials under the RAAF operating conditions. This led to closer cooperation between ARL and the RAAF who had given farsighted support to the fatigue research work from the beginning, and the stimulus and insight from the applied projects which had been so valuable to the research work was still available. However, it forced a

considerable tailing off in the DCA work. This was a considerable loss to ARL who had gained so much from this fruitful collaboration and it was also a loss to DCA who had relied on the R & D support of ARL in dealing with the major aircraft companies overseas. DCA and its successors subsequently sought DSTO assistance. A good example is when multiple small cracks (now referred to as Multiple Site Damage - MSD) were detected in several fuselage lap joints of an Australian regional carrier, several years before the infamous incident of the Aloha Airlines aircraft in 1989. The DSTO investigations included one of the first test programs on MSD [30].

Research work continued on the probabilistic aspects of fatigue [31]. A reliability approach to the fatigue of structures was developed which calculated the increasing risk of failure under the spectrum of service loads as the fatigue crack extends and reduces the residual strength of the structure; it took into account the variability in structural resistance and the variability in crack propagation rate of all structures in a fleet. The method enabled a safe operating life to be calculated, or if the structure is inspectable, it enabled inspection intervals to be calculated to maintain the required safety level by eliminating structures with detectable cracks. It was intended to be a quantitative approach which embraced both the safe-life and the fail-safe philosophies in the standing airworthiness specifications.

The earlier work on variability in fatigue life was also extended to modern ultra high strength steel structures using results from the tests on the large F-111 components referred to later. This development owed much to the earlier work on ultra high strength steels undertaken for DCA who were concerned about the performance of such steels in civil transport aircraft.

The investigation on flight loads continued with a cooperative research project on turbulence using high altitude data from the American Lockheed U-2 flights in Australia, low altitude data from a Mirage aircraft fitted with a gust probe developed by ARL and a network of anemometers mounted on towers at the Bald Hills station near Brisbane.
ARL had also developed a device for counting the fatigue damaging strain fluctuations in fatigue critical areas of an aircraft in service [32]. This instrument used a small magnetic core (computer type) memory for storing strain data from electrical resistance strain gauges in flight.

Basic research also continued on the fatigue of materials. Extensive information had been obtained on the fatigue performance of aluminium alloys 7178 and 2265 particularly as regards the effect of stress concentrations. This allowed methods to be developed for predicting the fatigue properties (in the form of an alternating versus mean stress (A–M) diagram, see Figure 12) of components from basic fatigue data on the material of construction.

A programme of research was also undertaken to investigate the effect of thermal cycles due to temperature gradients in modern high performance aircraft arising from flight at supersonic speed. A fundamental problem was to determine the temperature distribution in the structure and hence calculate the thermal stresses. In a cooperative programme with the University of NSW [34] basic research was carried out on heat transfer,

particularly the difficult problem of heat transfer by convection in a closed cavity typical of those in fabricated aircraft structures. The theoretical predictions obtained from a computer solution showed good agreement with experimental results obtained from a study of the convection currents in a small-scale model. Other important work from the University of NSW at this time was by W. Elber on the phenomenon of fatigue crack closure [35][36].

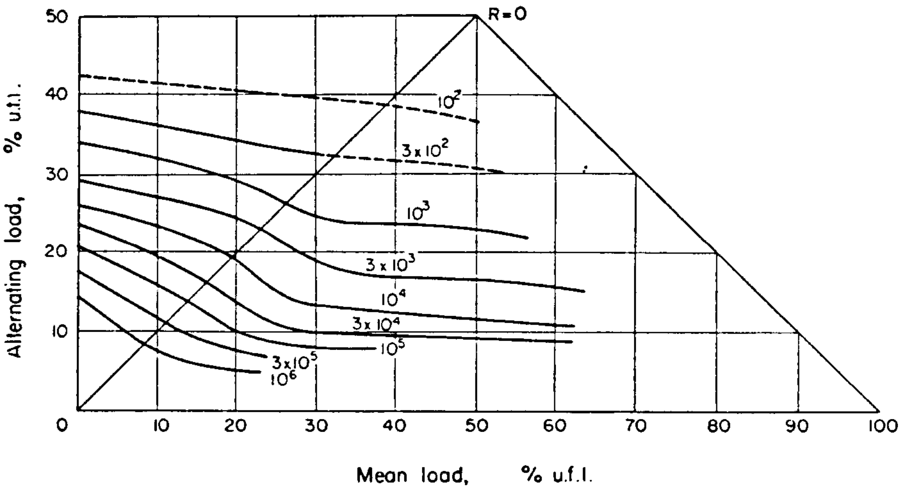

Figure 12: Example Mustang wing constant life curve (from [3])

The reliability approach referred to earlier was applied to establish an inspection procedure for the Aermacchi aircraft operating in Australia to enable the safe operating life to be extended. Fatigue data have been used from a FSFT carried out by Aermacchi together with results from a number of tests carried out by ARL on the critical component. The Aermacchi test was designed in a collaborative effort with Australia, and an ARL trained RAAF liaison officer was stationed in Italy for the duration of the program.

During this era the RAAF fleets all suffered fatigue problems to some degree and as a Defence Research Laboratory in the Australian Defence Scientific Service, ARL was committed to provide R & D support on a major scale. Fatigue life investigations were undertaken on Canberra, Hercules, Sabre, Neptune and Winjeel using fatigue data from the manufacturer or from ARL fatigue investigations and flight loads data from RAE fatigue meters [37] which were now coming into wide use in the RAAF fleets.

4.1 Vampire wing tests

For the Vampire aircraft a FSFT (Figure 13) was necessary to establish the fatigue performance of the locally produced wings under Australian flight conditions [38]. Twenty-one half wings were tested, using only two fuselages, and as a result an inspection and rework programme was developed which avoided replacement of wings which, since the RAAF had a fleet of 100 Vampires, was a major consideration. It was interesting to note that no fatigue failures were recorded in the wooden fuselages.

4.2 Cessna wing test

A fatigue test [33] was carried out on the Cessna 180 for the RAAF and DCA because it was typical of light aircraft used in the crop dusting and military role, in which there had been a spate of fatigue failures. There had also been a catastrophic accident to a Cessna in New Zealand caused by fatigue failure in the rear spar root and fatigue cracks had been found in similar locations in Cessna aircraft in other theatres.

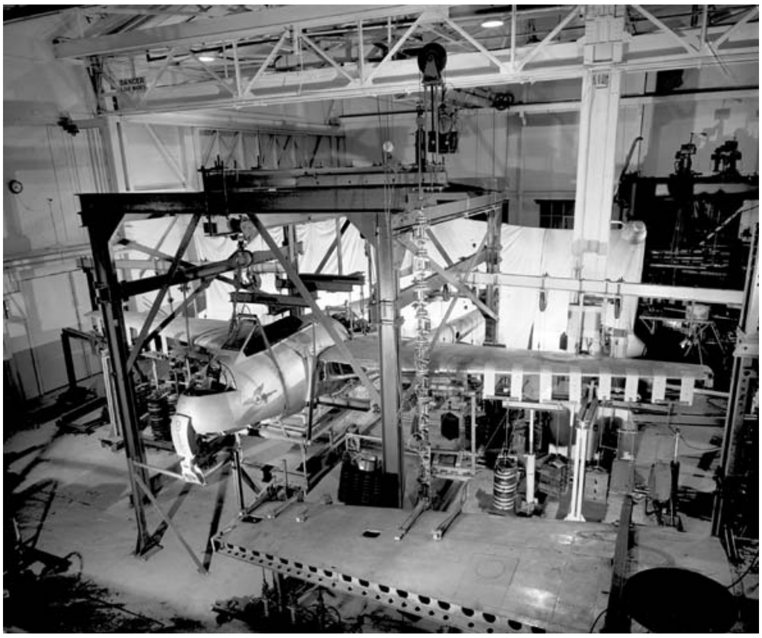

Figure 13: Vampire fatigue test rig (note the chains of notched specimens in front of the port wing)

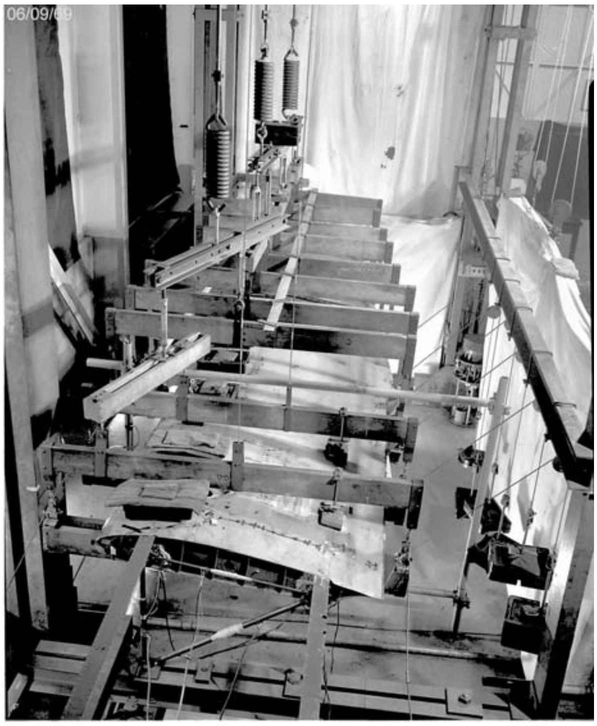

Figure 14: Cessna 180 wing test rig

4.3 Mirage wing test

A major fatigue investigation (see [39] for details) was undertaken on the Marcel-Dassault Mirage III 0 aircraft which was then in service with the RAAF. Since it was a supersonic aircraft operating in a new regime it was decided to first undertake a comprehensive flight investigation to determine the temperatures and stresses in various flight roles. Preliminary fatigue life estimates for the RAAF conditions were then made using the flight load information obtained in the ARL flight tests with fatigue data from the simplified tests carried out by the aircraft manufacturer for their own requirements. These tests involved not only dissimilar operating conditions, but also a much shorter operating life than that envisaged by the RAAF, defining the need for the FSFT.

The test load sequence was designed to closely represent the complex load sequence experienced in RAAF service by constructing it from actual flight records obtained from squadron operations. The continuous load sequence was put onto magnetic tape and applied to the structure with and ARL developed instrumentation and control system (see [40][41] for details). The testing rig (Figure 15) was the first multi-channel test to use closed loop servo hydraulic controlled loading at ARL. It included an ARL closed loop servo controller and an ARL safety unload servo hydraulic valve pack, developments that became the foundation of testing systems at ARL right through to the present day. Because of its high wing loading and heavy gauge skin, bonded pads were used to attach actuators to the wing skin through the whiffle tree lever system.

This test also included undercarriage spring-back loads as well as fuel tank pressurisation. The various fatigue failures that had appeared were monitored until final failure occurred.

This has enabled a longer life to be established for the wing and that enabled the aircraft to remain in service with the RAAF for many years.

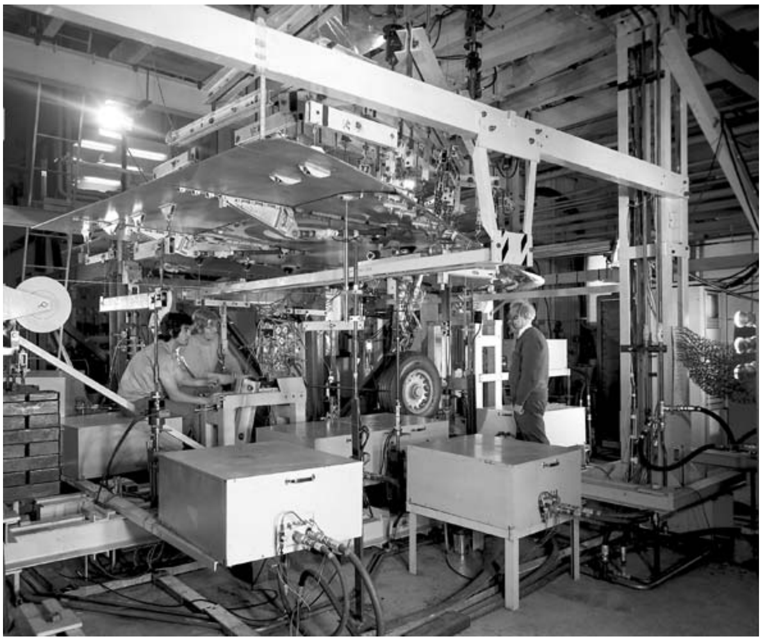

Figure 15: Mirage wing test rig

4.4 F-111 component tests

Perhaps the most important and difficult fatigue investigation undertaken for the RAAF was on the F-111. The new design features and the use of ultra high strength D6ac steel on which little experience existed, presented General Dynamics with major problems in fatigue design and a number of early failures occurred in the FSFTs carried out at Fort Worth. The early work on similar high strength steel for DCA had provided ARL with valuable background and ARL scientists were called in by the Department of Air to visit Fort Worth for discussions regarding the fatigue performance of the F-111 under Australian conditions and to advise the Australian Air Board.

Twelve large ultra-high strength steel components representative of the F-111 construction were given to ARL for fatigue testing by the USAF in response to representations by Australian scientists on the insufficient knowledge and experience available on the fatigue performance of this new type of construction. The testing of these large high strength components required a 500,000 lb. "Humpheries" fatigue-testing machine (Figure 16) which was designed and built by ARL and the engineering establishments within the Department of Supply to a very short time scale. The tests soon revealed a new and unsuspected problem in that it was found that various fluids used in the machining and production of the components had a corrosive influence which greatly affected the fatigue crack propagation rate. These findings were soon confirmed by the appearance of similar problems in the F-111 aircraft going into service in the United States.

The ARL Scientific Advisory Panel, headed by the Chief Superintendent made a number of recommendations to the Air Board concerning the fatigue life substantiation of the F-111

under Australian conditions. The F-111 programme was a major activity in ARL for over 2 years and a number of ARL scientists visited the United States for periods of up to several months for discussions and investigations at General Dynamics, and the Materials Laboratory of the USAF Aeronautical Systems Division, Dayton, Ohio. Australian scientists accompanied the mission in 1969 by Sir Henry Bland and the Chief Defence Scientist, Mr. H.A. Wills and also the mission by Mr. Fraser, Minister for Defence in 1970 which visited the United States to negotiate on conditions for acceptance of the Australian aircraft. Mr A. Patching of ARL was stationed at Fort Worth for two years as a member of the Structural Monitoring Team and was invited to be a member of the USAF Scientific Advisory Board ad hoc committee on the F-111. The committee consisted of civilians chaired by Professor H. Ashley from Caltech.

It is perhaps appropriate that ARL which originally took the lead in investigating fatigue in aircraft structures should have played an important part in the solution of one of the most difficult fatigue problems of the day, and that as Chief Defence Scientist Mr. Wills himself was a key figure in the negotiations leading to the successful introduction of the F-111 into RAAF service. It also forced recognition of the fact that undetected structural cracking did not equate to flaw-free structure. The subsequent USAF requirement to design aircraft structures against the assumption of existing cracks in all fatigue-critical locations (damage tolerant design), to achieve service lives no less satisfactory than before, is one of the more remarkable developments of recent times.

Figure 16: F-111 component test in the Humphries test machine

5. Period 1975-1990:

In line with the more specifically Defence-oriented role of the Laboratories, four major military fatigue programmes were carried out in the 1970's [42]. Following the detection of cracking in service of Aermacchi centre-section booms, a laboratory test programme showed that crack growth rates from critical holes were low and could be readily monitored in service. Safety-by-inspection management of these aircraft was subsequently put into effect, the Aermacchi becoming the first RAAF aircraft to be operated with known fatigue cracks in primary structure. The first ARL flight-by-flight fatigue test was carried out on a Mirage wing in 1973/4, and revealed the fatigue sensitive nature of the main spar wing root region: this test was done in close collaboration with the Swiss who were testing a complete airframe at this time. In 1976, a flight-by-flight fatigue test began on a Nomad airframe: this test was undertaken in the Wing Bay by Aerospace Technologies of Australia (formerly Government Aircraft Factories – GAF, currently Boeing Australia).

Following a RAAF requirement for an increased service life for its Mirage fleet, a substantial laboratory programme, occupying some years, was initiated to study the options available. This work demonstrated that interference fitting of bushes in some fatigue-critical holes and the cold-expansion of others was practicable and provided the required life extension: the modifications were subsequently applied to service aircraft.

An ARL proposal for developing a strain-range-pair counting device for monitoring aircraft loads in the early 1970's led to the development of the Aircraft Fatigue Data Analysis System (AFDAS) [32] being fitted to RAAF aircraft. The system provided estimates of fatigue damage being accumulated by individual aircraft, thereby allowing appropriate fleet management options to be exercised as required.

During this period reliability theory was applied to the F-111 aircraft with a view to defining well-founded inspection intervals. Studies to establish fatigue crack propagation rate data under various loading and environment conditions became routine, and the proliferation of suitable applications of composite repairs led to many laboratory verification programmes. By the end of the decade RAAF F-111C, Aermacchi and Mirage aircraft were all being operated, at least in part, on a safety-by-inspection basis – this represented a significant change from the previous long-standing safe-life tradition.

Following a redesign of the centre-section joint of the main spar of the CT-4 trainer aircraft, a comprehensive flight strain survey and a fatigue test were carried out. The test article survived over 25,000 service loading flights following which the severity of loading was increased to precipitate failure. This occurred after a further 25,000 flights. The test demonstrated the very robust fatigue capability of this aircraft. The Nomad fatigue test reached over 300,000 flights without failure of the main wing. Although several failures occurred (mainly in the strut and stub-wing), the test proved valuable for verifying refurbishment options and NDI developments.

Serious fatigue cracking of the wing pivot fitting of F-111 aircraft led to a significant programme for the design and application of boron fibre reinforcement doublers to this

region [43]. Unlike most previous applications, which were to thin-gauge regions, this application was to a primary load carrying region of ultra-high-strength steel in which stress levels are very high. The usefulness of fibre composites for reinforcement and repair of structural components continued to increase. Durability trials under fatigue loading in a tropical environment were carried out in Queensland on a long-term basis, and the search for better and more durable bonding materials and processes continued as a matter of course.

Loading sequence interaction effects were investigated and the principles for reconstituting loading sequences from condensed (range-pair) data established. Theoretical and experimental studies on interference fitting and the cold-expansion of holes established clearly the means by which these processes achieve their quite remarkable effect in increasing fatigue life. Global stress and displacement measuring techniques were developed to provide indispensable aids to fatigue understanding.

5.1 The Government Aircraft Factories Nomad test

In 1976, a flight-by-flight fatigue test began at ARL (Figure 17) on a Nomad series N22/24 part fuselage and wings by the GAF [44]. This main wing in the article survived over 300,000 simulated flights without failure [45]. Details of the ARL computer controlled servo-valve module, similar to that used in the Mirage wing test, as well as the whiffle tree layout is given in [46].

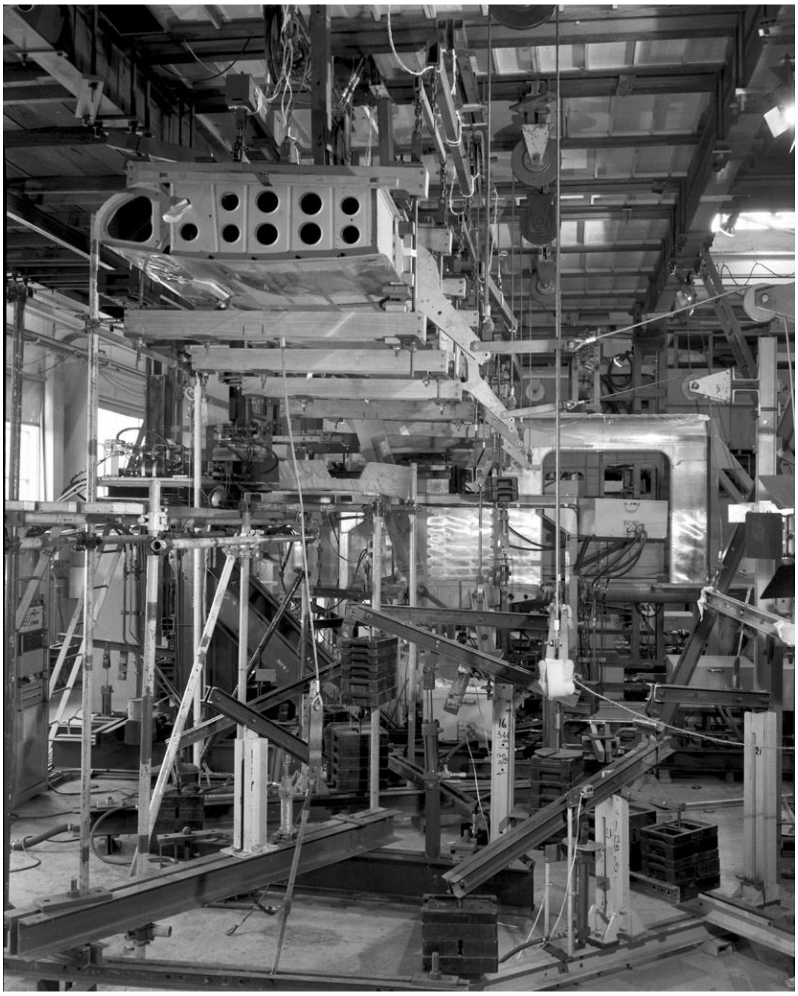

Figure 17: GAF Nomad fatigue test article in the ARL "Wing Bay"

5.2 The CT4 test

In July 1972 the Australian Government announced, that approval had been given to the selection of the Aero Engine Services Ltd New Zealand – subsequently Pacific Aerospace Corporation – CT4 air-trainer (a derivative of the Australian designed Millicer-Victa Aircruiser). The aircraft was designed to conform to FAR 23 Civil Airworthiness Requirements, based on safe-life concepts with an approximate all up weight of 1179 kg (2600 lb). This aircraft was recommended as a suitable replacement for the "Winjeel" aircraft then in service as the RAAF basic trainer. The Minister of Defence also stated that the ARL, in collaboration with the RAAF, would conduct a fatigue test programme on the complete CT4 airframe, and that this test would be the basis of safe life certification. This decision was no doubt influenced by the low fatigue life estimates and the wide variation of estimates between independent authorities [47]. According to the then current RAAF Specification Engineering No. AC 167, the air-trainer was required to have a service fatigue life of at least 5000 hours in the training role.

The Department of Aviation required that the fatigue test programme conducted on a complete CT4 airframe under a loading spectrum produced by the RAAF be completed within the first two years of RAAF service, i.e. by early 1976. However, it was not until March 1979 that the Air Force Office finally gave approval for the test and supporting flight trials to proceed.

A test rig incorporating control and loading equipment developed at ARL was designed and manufactured by the GAF and installed at ARL during 1982. Following a rig-commissioning period the test commenced in June 1983 and the first major structural failure occurred in August 1984. The fatigue test rig (Figure 18) comprised six steel columns supporting crossbeams in the form of a grid, from which the test airframe was suspended and on to which the loading jacks and associated mechanisms were mounted. Electro-hydraulic servo controlled loading jacks were used to apply the test loads at seven locations. Representative distribution of wing and horizontal tailplane loads was achieved through proportioned loading beams (whiffle trees). Direct application of loading was employed in the case of the undercarriage vertical, side, and drag loads, and also fin loads. The engine-mounting frame was used only to react fuselage loads and was not subjected to representative inertia and torque loading. Although this test rig arrangement is now common practice, several noteworthy advances were incorporated at the time including the ARL developed miniaturised manifold controlled servo-jack control system [47], and the use of a parallelogram linkage arrangement between the upper and lower whiffle trees so that the overhead actuators could apply both up and down loads.

Figure 18: The CT4 test rig

The flight-by-flight spectrum considered load exceedance data for the training bases (then Point Cook and East Sale RAAF bases) as well as gust load exceedances, landing counts data and rare load occurrences. The sequence representing this spectrum was applied for 521 programmes (6248 turning points) or 20470 equivalent service hours (ESH) and when added to the previous sequences represented a total life of 25460 ESH or 648 programmes.

This was slightly greater than that required by the test specification of attaining test life equivalent to 5000 hours after the application of a scatter factor of 5. Upon reaching this test milestone it was decided to accelerate fatigue damage accumulation by the application of a more severe load spectrum which was applied until the first major failure occurred at a total life of 51141 ESH. The initial mainplane failure occurred in the port wing main spar tension boom at station 641 mm (25.25 inches). The general mode of failure, the fracture surface and Quantitative Fractography (QF) derived crack growth curve are shown in Figure 19. The fractographic examination of the failure surfaces indicated that cracking initiated in the T-section extrusion close to the start of the test.

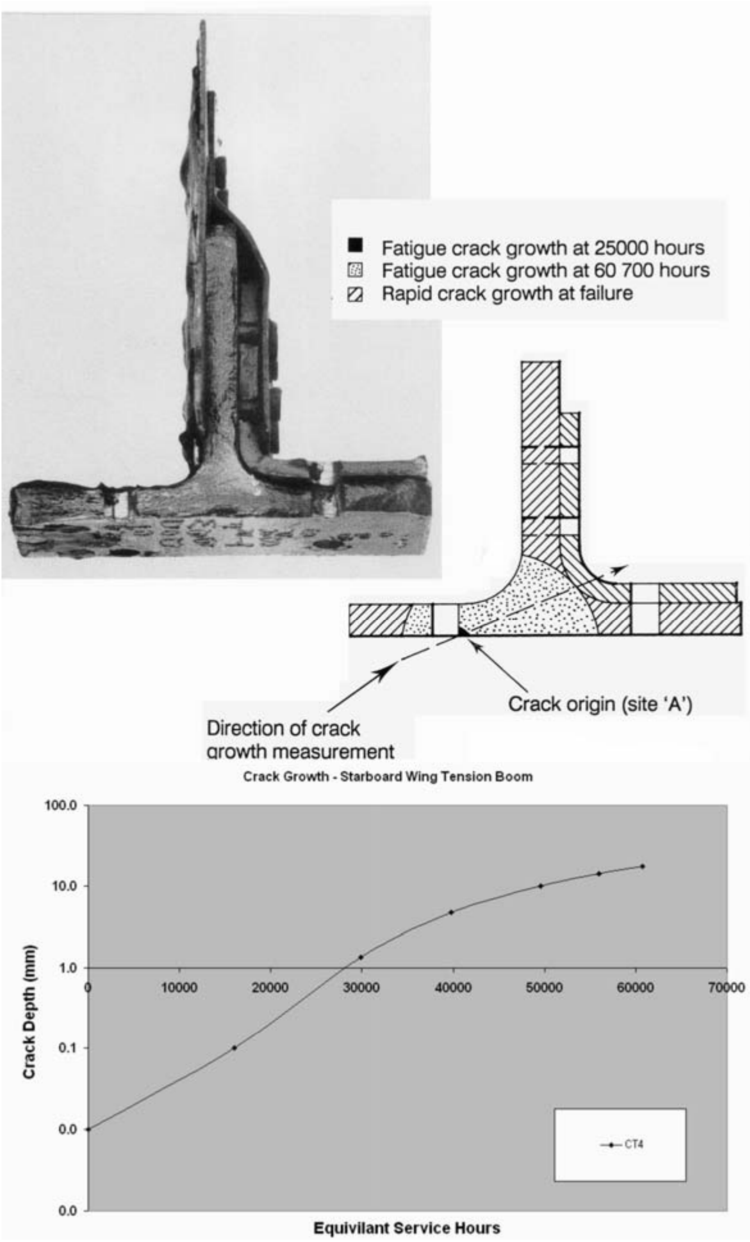

Figure 19: CT4 starboard wing main spar tension boom, general mode of failure and QF derived crack growth curve.

Prior to commencement of testing it became apparent from flight trials strain data that the majority of high strain measurements recorded in the empennage developed from buffeting during spin recovery manoeuvres which caused resonance. It was decided therefore to include dynamic loads at resonant frequency in the empennage test loading. However, because of many technical difficulties involved in combining resonant loads with manoeuvre loads in a test rig it was decided to separate the two types of loading by applying loads of up to 1 Hz frequency in the main fatigue test rig and the resonant loads of approximately 9 Hz in a specialised dynamic loading rig [47][48].

Figure 20: CT4 horizontal tail resonance test rig

Figure 21: CT4 vertical tail resonance test rig

Tailplane loading up to the first major mainplane failure consisted of low magnitude symmetrical airframe balancing loads only. Following the first major failure the empennage was disconnected from the airframe and installed in the dynamic rig for resonant loading (Figure 20 and Figure 21). It was then reassembled onto the fuselage in the fatigue test rig and subjected to symmetric and asymmetric manoeuvre, followed by periods of buffet loads for the remainder of the airframe test. This experience with dynamic testing was to influence the direction of the F/A-18 aft fuselage and empennage test (see Section 6.1).

5.3 The Pilatus PC9 test

In 1986 the RAAF purchased 67 Pilatus PC9 trainer aircraft. A number of modifications were specified, of which the main structural modifications were provision for low-pressure tyres for grass field operation and inclusion of wet assembly at faying surfaces. The particular version of the aircraft supplied by Pilatus was designated the PC9/A. The aircraft was certified to comply with US FAR Part 23 in both the Aerobatic and Utility categories.

Pilatus warranted the PC9/A aircraft to have a fatigue life of 8,000 hours when operated within the bounds of an agreed Nz exceedence spectrum, which became known as the New Basic Trainer Aircraft warranty spectrum.

As no FSFT of a PC9, or its predecessor the PC7, had been carried out, such a test was viewed by the RAAF as essential to establish baseline information for long-term fatigue management of the PC9/A fleet. The desire to assess the structural modifications unique to the RAAF fleet strengthened the argument for a FSFT. An additional airframe (number 68 on the production line) was procured as a fatigue test article.

The (now) AMRL[1] was requested by the RAAF to undertake the fatigue test. A test specification supplied by the RAAF limited the number of active loading channels to approximately 24. The specification defined the designated test structure as the wings, main landing gear support structure, tailplane, elevator, fin, fuselage and engine-mounting support structure. Control surfaces such as aileron, rudder, flaps and speed brake were not part of the designated test structure. It was also specified that the test load sequence would be based on flight-recorded data gathered from an extensive flight test program undertaken by the RAAF. These data were used for developing test loads representing general manoeuvre flying. Spin loads were derived from special spin flights because it was observed that spinning caused vibration from suspected buffet loading of the empennage. The overall arrangement of the test is shown in Figure 22. This was the first FSFT to be conducted in the new Structural Test Laboratory at Fishermans Bend.

The overall load sequence was applied in blocks representing 250 Simulated Flight Hours (SFH) of operation comprising 197 general manoeuvre flights, made up of a variety of sortie types. The Nz exceedence counts associated with the sequence was chosen to match the average usage for the fleet of aircraft for the period 1990 and 1991. Test rig commissioning began in January 1996 and 50,000 SFH were achieved by February 1999 [49]. Continuation testing proceeded until 67,000 SFH (July 1999), at which time the wing main spar failed. The test rig was then re-configured to allowing continuation testing of the empennage and aft fuselage. This testing was completed without major structural failure at 100,000 SFH (January 2000). A residual strength test of the fin and tailplane was then conducted, followed by full teardown inspection.

The test loading system was arranged such that the central lower region of the wing was accessible while the test was running, and test operators and inspection staff performed daily inspections of this region. These inspections proved valuable, as several cracks were detected. Pilatus Ltd also became involved in the program during the test conduct phase mainly through the design of repair schemes.

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[1] ARL was merged with the Materials Research Laboratory in 1994.

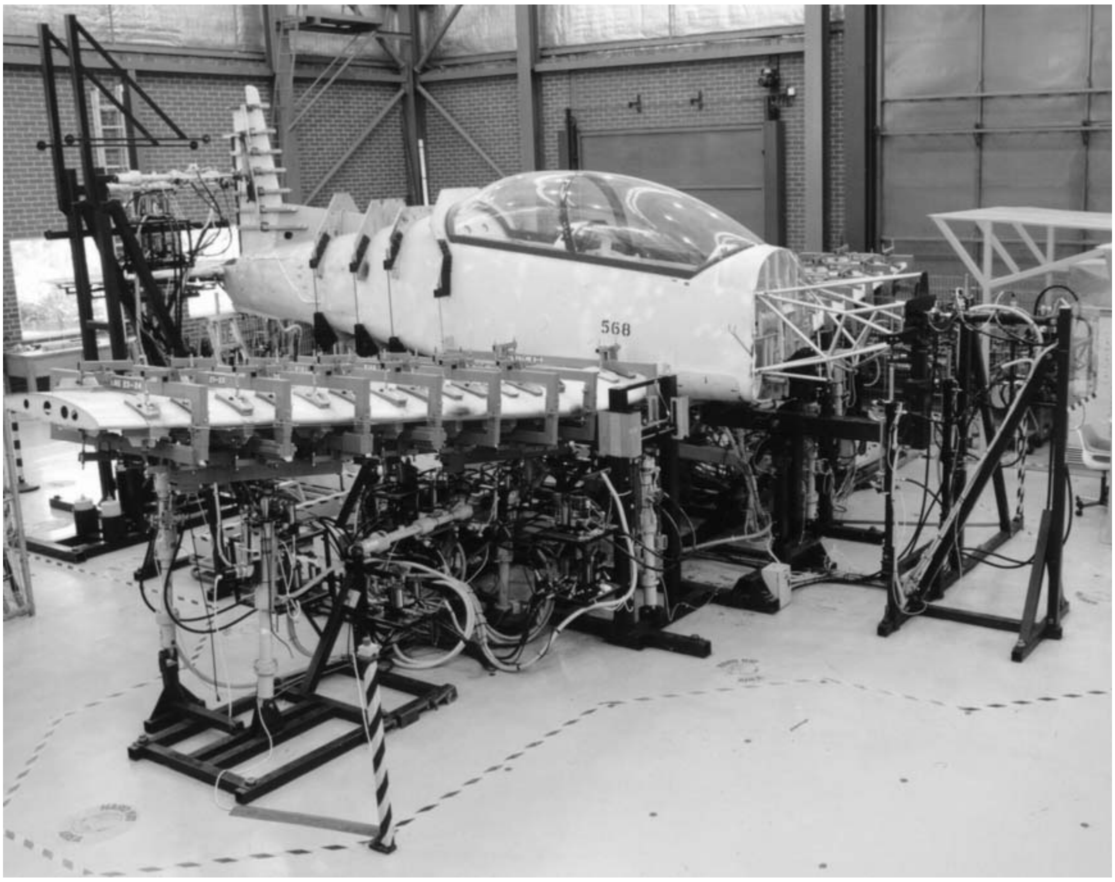


Figure 22: PC9/A fatigue test

6. 1990 – present

The period from 1990 to the present has been a very busy time for the Fishermans Bend fatigue laboratories. The innovative F/A-18 aft fuselage and empennage FSFT placed considerable strains on available resources but, after a development period of approximately ten years, provided ample data to allow the fleet to be operated to a planned withdrawal date. The program also won the prestigious Von Karman Award at the 2002 International Aerospace Congress in Canada. The aging F-111 aircraft provided considerable structural integrity management difficulty. Along with structural optimisation programs, DSTO was involved in the conduct of three separate FSFTs of F-111 wings.

DSTO was also involved in the P-3 Service Life Assessment Program (P-3 SLAP) program of FSFTs and analyses to determine the structural life of the P-3C aircraft. The program was a collaborative effort between the USN, RAAF, CF, Royal Netherlands Navy and Lockheed-Martin (L-M) Marietta USA.

The new Original Equipment Manufacturer (OEM) through life logistic management concept also impacted DSTO through the RAAF's purchase of the Hawk Mk127. The purchase contract specified the provision of capability (so many flying hours per year etc)

rather than a fixed number of aircraft. It was envisaged that the OEM was to be responsible for the through life structural integrity support of the aircraft. The Mk127 was significantly different from other variants of the aircraft and so the OEM (British Aerospace (BAe) Systems) defined the requirement for a FSFT to be conducted in the UK. However due to the strong support from the RAAF's Directorate General Technical Airworthiness the Commonwealth added additional funds to the contract to support the conduct of the FSFT in Australia. With the anticipated wind down in F/A-18, P-3C and F-111 testing, DSTO was keen to maintain a capability in FSFT, and the Hawk Mk127 provided such an opportunity. Amongst other factors, this allows young airworthiness practitioners to develop the requisite assessment skills through the real-world experience gained through the conduct of FSFTs. Thus DSTO found itself in the interesting situation of conducting a FSFT under contract to, and specification of, BAe Systems whilst setting up "Chinese walls" to perform its corporate governance role in advising the RAAF.

Possibly the most challenging fatigue investigation ever undertaken by DSTO was related to the fatigue failure of an Aermacchi MB326H lead-in-fighter aircraft in November 1990 [50]. Until 1990 the structural integrity of the fleet had been founded on a safe-life approach, with the aircraft life deemed expended on achieving a portion of the life demonstrated in the Aermacchi FSFT conducted in 1974/75. After centre section replacement, the next major failure occurred in the lower wing spar following which the test was terminated. In the early 1980s, the RAAF conducted a Life-Of-Type EXtension (LOTEX) program in which various modifications were performed. The modifications were performed by a local contractor and not Aermacchi. Unfortunately in hindsight no FSFT of the modified wings was undertaken, and as a result, the RAAF no longer had knowledge of the second most critical fatigue locations in the wings. After the in-flight separation of a wing during air combat manoeuvring, later estimated to involve a 6.5g vertical acceleration, at a service life substantially lower than the promulgated life limit, the entire fleet was grounded. The parts recovered from the sea identified a fatigue failure in the port lower wing spar cap. Subsequent investigation revealed that the problem stemmed from poor drilling of the web-to-spar cap fasteners during the LOTEX program. Teardown inspections conducted at DSTO of other similarly high time aircraft proved that the incident was not isolated and an NDI program was developed. A new safe-life was also set that represented approximately half the previously promulgated safe life. The fleet recovery program was also extended to include the condition of other significant structural components. Crucial to the effectiveness of the program was the accelerated fatigue testing and teardown of ex-service tailplanes and extensive inspections of fuselages. The investigations indicated widespread fatigue and corrosion related problems throughout the fleet. The MB326 fleet was retired with the introduction of the Hawk 127 in 2000.

The probabilistic fatigue approach pioneered by ARL in the 1950's was further refined and became known more commonly in this period as the risk and reliability approach. It has been used in various investigations for the RAAF's Boeing 707 fleet [51], the F-111 [52] and F/A-18 [76]. In the case of the F/A-18 the approach was expanded to include the effects of variations in initial flaw sizes, fracture toughness, crack growth rate and the accuracy of the fleet fatigue monitoring system [76]. That same investigation also identified several limitations when using aircraft design standards for through life structural integrity

management. A new approach to the interpretation of FSFT and life estimation was proposed [77] which relied on the results of QF analyses of fatigue cracks excised from the test article as well as fatigue coupon tests conducted for representative material and surface finish as well as the same FSFT spectrum. It was shown that the stresses at critical locations could be accurately estimated by comparing the growth rates of the cracks from the FSFT with those from the coupon tests. In a key departure from traditional certification FSFT it was recommended that FSFTs should be conducted until the extent of damage was uneconomical to repair. Learning from the Aermacchi experience, the aim was to discover as many critical locations as possible and provide crack growth data for these locations. This information could form the basis of the probabilistic life assessment as well as providing the option for life extension if required.

This period also saw significant activities in the area of fatigue prediction. Through DSTO's experience in interpreting fatigue cracks through QF, even the most widely accepted of crack laws, namely the Paris Law (Equation 1) [78], was questioned for general application.

$$da/dN = C\Delta K^m \qquad \text{Equation 1}$$

Surprisingly these observations were consistent with the early ARL work of A.K. Head [12]. Frost and Dugdale [79][80], using Head's observation of self-similar crack growth, expanded Head's law and reported that crack growth under constant amplitude loading could be described via a simple log linear relationship, viz:

$$Ln(a) = \beta N + a_o \quad \text{or} \quad a = a_o e^{\beta N} \qquad \text{Equation 2}$$

where N is the "fatigue life", β is a parameter that is geometry and load spectrum dependent, *a* is the crack depth at time N, a_0 is the initial flaw-like size (depth of the crack at the start of loading) and:

$$\beta = f(\sigma) \qquad \text{Equation 3}$$

for constant amplitude loading Frost and Dugdale [79][80] found that β could be expressed as:

$$\beta = \lambda\, (\Delta\sigma)^3 \qquad \text{Equation 4}$$

In this context, researchers at DSTO have for many years observed, in a wealth of experimental data (summarised in [81]), that exponential growth rates are a good approximation for most of the life of cracks grown in service or from FSFT, and coupon tests under service spectra (for example see *Figure 25*). These data generally show that cracks grown under typical service spectra produce crack growth histories that can be expressed to a good approximation by Equation 2. This relationship appears to be valid whenever the stress intensity factor is a linear function of $\sqrt{a}$. Other researchers have also commented upon the apparent exponential rate of crack growth at small crack lengths and confirmed that Equation 2 holds for both micro- and macro- crack growth.

While this relationship appears to be applicable over most of the life of typical cracks, growth rate acceleration towards the end of life can cause an upward trend in the crack growth acceleration rate. From QF observation of many cracks this appears to be accounted for by the onset of tearing, changes in geometry and the coalescence of other cracks growing nearby to form a larger crack – i.e. a linear function of $\sqrt{a}$ no longer applies. The important point is that this deviation from exponential growth only accounts for a small fraction of the total crack life where load shedding is not involved.

As a consequence of the relationship in Equation 2 and numerous crack growth measurements taken of coupons loaded at several different stress levels, DSTO have shown that for aluminium alloy 7050-T7451 Equation 4 also holds to a good approximation (see [81]). These types of relationships have been used (eg [76][77]) to predict the growth acceleration rate (β in Equation 2) of cracks at stress levels other than those tested, while the effective crack like nature of the initial discontinuity from which a crack might grow, for a specific surface, has been established by back projecting to time zero from the measured results of test or service cracks using Equation 1 (see for example [82]). This work suggested a crack growth law of the form:

$$da/dN = \underline{C}a^{1-m^*/2} (\Delta K)^{m^*-} \qquad \text{Equation 5}$$

that resembles the Paris growth law, and where setting m* = 3 yields Equation 2. Also in related investigations (for example [83]) it was shown that Equation 2 yielded good results if the co-efficient was derived directly from spectrum crack growth data.

Other research was directed at addressing the problem of fatigue life estimation of components experiencing high levels of plastic deformation, such as the F-111 wing pivot fitting during periodic cold proof load tests (CPLT). Initially, this type of fatigue crack growth analyses involving notch plasticity were carried out using the software METLIFE from Lockheed Martin, but the following issues with METLIFE motivated the development of the DSTO CGAP software [84]; numerical problems, no transient behaviours modelled such as plastic strain ratchetting and mean stress relaxation, and no plasticity-induced crack closure considered. CGAP addressed these, and preliminary results showed an encouraging trend compared to experimental data. CGAP is MS Windows-based with a user-friendly graphical user interface, and it provides a database capability for the management of geometry, material, load and solution information. Apart from the deterministic crack growth, it also provides a capability in probabilistic crack growth analysis based on the Monte Carlo method, with randomised initial crack size, crack growth rate and the peak spectrum stress.

6.1 The F/A-18 aft fuselage test

The RAAF and Canadian Forces (CF) operate the Boeing F/A-18 in a non-aircraft carrier role. This has resulted in usage quite different to that specified by the principal user, the United States Navy (USN) for airframe certification. Given similarities in operational usage, airframe configuration and airframe management philosophies, the RAAF and CF embarked on a significant collaborative project known as the International Follow-On

Structural Test Project (IFOSTP). IFOSTP consisted of three separate FSFTs supported by flight trials and load development programs (see [53]). Both RAAF Aircraft Research and Development Unit (ARDU) and CF Aircraft Evaluation and Test Establishment conducted the flight trials. In arguably what was the most challenging fatigue test ever undertaken by DSTO, the Australian portion of IFOSTP consisted of a unique full-scale aft fuselage and empennage fatigue test (known as FT46) which combined buffet induced dynamic loading with manoeuvre loading to reproduce the flight loading conditions experienced by an F/A-18 aircraft under normal flight operations (see [54] to [65]). Loading spectra were derived (from a combination of flight trials data and data recorded on-board each aircraft) to be representative of typical RAAF and CF usage. FT46 was particularly challenging because of the loading environment the aft fuselage and empennage experienced in service.

The F/A-18 is a highly manoeuvrable, high performance fighter/attack aircraft. The inner wing leading edge extension (LEX) provides fuselage lift enabling it to achieve angles of attack (AOA) in excess of 60 degrees. The twin vertical tails canted slightly outward exploit the high energy vortices generated by each LEX to provide good directional stability at these high AOA conditions. Unfortunately, these vortices break down at high AOA, buffeting the empennage and exciting the vibration modes of the empennage structure. This buffet phenomenon results in severe empennage dynamic loading, as indicated by the high acceleration levels measured at the aft tip accelerometers of each vertical tail (approximately ±550*g*) and horizontal stabilator (approximately ±150*g*). This empennage vibration also excites engine and other aft fuselage resonant dynamic response causing high stress levels in many structural components. The majority of the fatigue damage imparted to the aft fuselage and empennage was due to this severe buffet dynamic loading, which could occur simultaneously with high manoeuvre loading. There is a synergistic interaction between this quasi-static manoeuvre loading and the higher frequency buffet loading with respect to fatigue damage. This interaction, along with the large number of dynamic load cycles caused by the significant amount of time the aircraft spent above 10 degrees AOA, dictated that the aft fuselage fatigue test employed a test approach which would simulate as realistically as possible the combined manoeuvre and dynamic loading experienced in flight.

The primary objective of the loads development process was to ensure that the test article was loaded to match its dynamic response as closely as possible to that of an aircraft in flight. This was achieved by having a manoeuvre loading system that would not significantly affect the dynamic characteristics of the structure. Typical fatigue test loading systems use hydraulic actuators, loading beams and pads, and cables. Such a system was unacceptable for FT46 testing, as it would add too much mass, stiffness and damping to the structure and alter the test article's dynamic characteristics adversely. To circumvent this, DSTO developed a pneumatic loading system, rather than the more conventional hydraulic loading system, to apply the aircraft distributed aerodynamic and manoeuvre-induced inertial loads. The buffet-induced dynamic inertial loads as experienced in flight were applied using a multi-channel vibration control system and high-powered, high displacement electromagnetic shakers.

FT46 completed 23,090.2 SFH using two spectra representative of distinct periods of RAAF and CF usage in June 2002. The first test phase applied loading representative of loading experienced prior to the installation of LEX fences. The LEX fences were retrofitted by the OEM to alter the aerodynamics of the leading edge extension vortices to reduce the empennage buffet dynamic response. The second test phase of loading represented service loading after the installation of LEX fences to the aircraft. In the third phase of the FT46 test program, residual strength testing (RST) was completed and the test article was torn down for a final inspection (completed in December 2004).

6.1.1 Test System

The test system was developed during a five-year development program utilising an early centre/aft static test fuselage provided by the USN. The essence of the manoeuvre load application system is a unique rolling sleeve pneumatic actuator ("airbag") that had soft spring stiffness and low mass. Using this system, the distributed manoeuvre loads were applied without significantly affecting the dynamic characteristics of the empennage structure. Several opposing air springs were used on each empennage surface to allow bi-directional loading. Hydraulic actuators were used to apply engine thrust loads, speed brake loads and in-plane stabilator loads required to obtain correct stabilator root manoeuvre loading for the fixed 12 degree leading edge down stabilator position. Concurrently, electromagnetic shakers applied the dynamic loading, while a hydraulic active reaction control system maintained almost zero displacement of the test article empennage to minimise shaker stroke requirements during high manoeuvre loading. Using this approach the significant number of dynamic cycles occurring over the service life of an aircraft was economically applied to a test article in real time.

The successful development of a combined closed loop operation of the air springs, electromagnetic shakers and hydraulic actuators was critical to the program. The controller controlled 65 inter-dependent load channels such that the manoeuvre loads were controlled to within 2% of the required spectrum loads. The final test arrangement is shown in Figure 23 and Figure 24. The control system also incorporated significant redundancy, safety systems and fault sensing; compensated for multi-channel interaction; permitted additional "ad-hoc" data acquisition; managed three fatigue test data acquisition systems; allowed data capture during unexpected shutdowns; created test controller databases/logs and managed the automated data archives used to store all fatigue test data (approximately 30 GB per test block).

A large reaction frame was built for the fin and stabilator manoeuvre loading actuators. A separate portal frame was built to support the vertical fin dynamic shaker system. All frames were attached to the laboratory strong floor. Because the test article was generally inaccessible for inspection when it was installed in the rig, a test article removal/installation rail system was designed to allow the test article to be rolled out of the rig for inspection. This provided a test article that was uninhibited by whiffle trees or loading pads during inspection. The actuators used to apply engine loading and fuselage reaction loading were installed on a removable trolley so they could be removed before the test article was removed.

The test article was instrumented with over 1500 strain gauges and load bridges and as many as 60 accelerometers. Response measurements from these channels as well as command and control channels were recorded on three data acquisition systems: the Buffet Data Acquisition System (BDAS), used to record 112 channels at a sample rate of 606.06 Hz; the Manoeuvre Data Acquisition System (MDAS); used to record 490 channels at each applied load line and; the Control Data Acquisition System (CDAS), used to record various control parameters. The data collected by these systems were synchronised and managed by in-house developed data acquisition and post processing software. At the end of the test it was estimated that if the data CDs (without cases) were stacked on top of each other the column would be approximately 9.3m high.

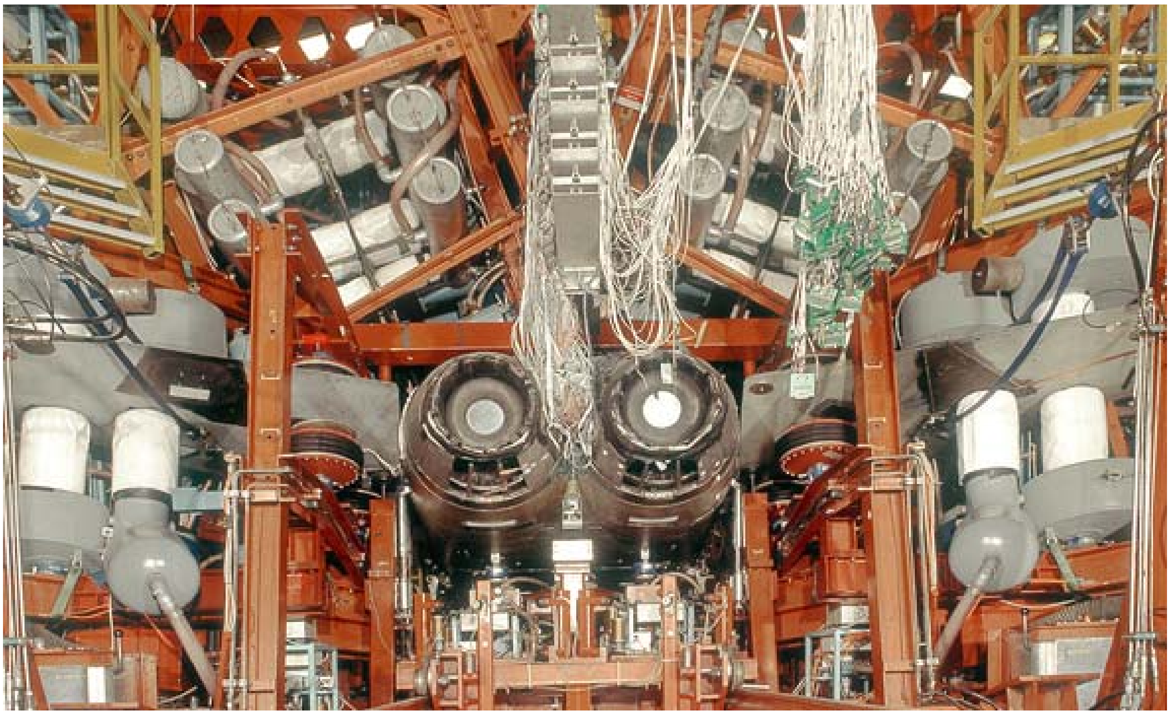


Figure 23: FT46 in test rig (rear view)

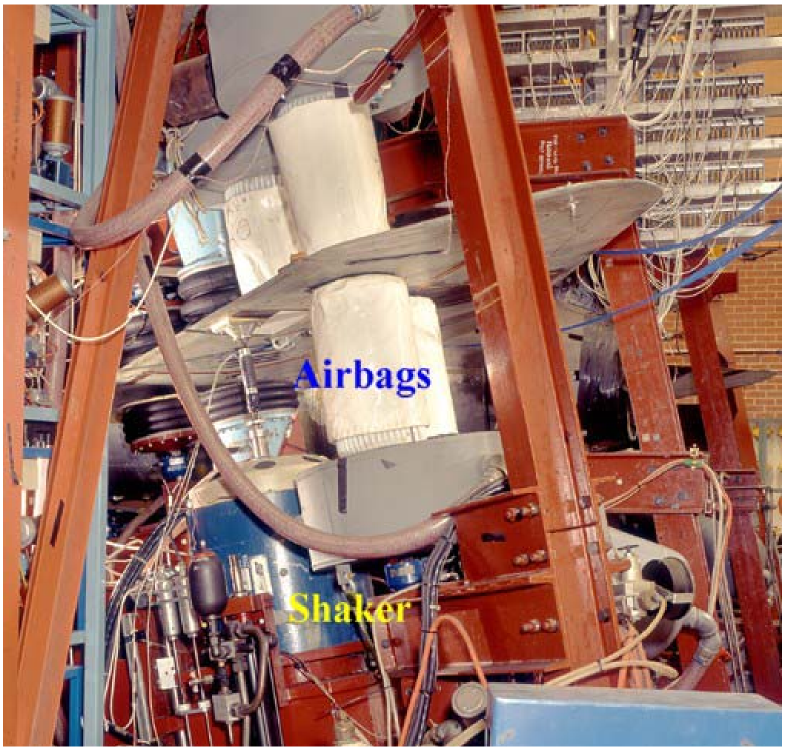
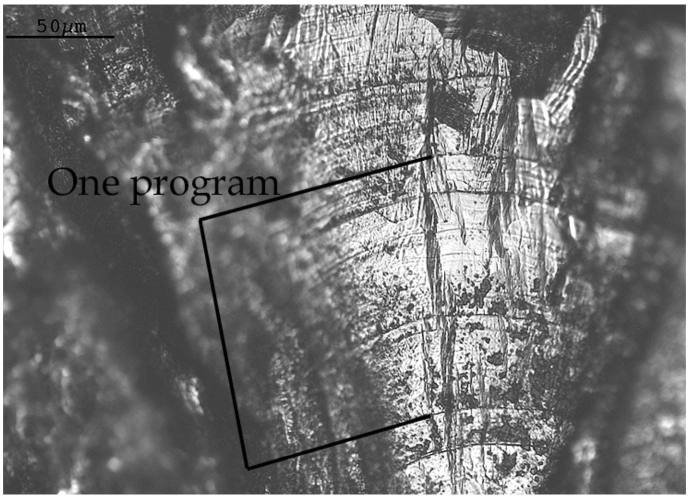


Figure 24: Detail of airbags and electromagnetic shaker in place on the horizontal tail

6.1.2 Results

To assist the analysis of cracking found during the FT46 testing DSTO/RAAF policy was to remove cracking intact if possible and use QF to determine the initiating discontinuity and the crack growth rate where possible. A typical result is shown in *Figure 25*.



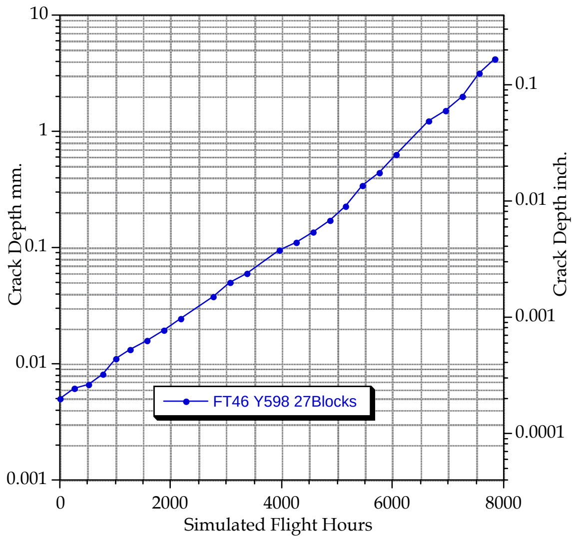


Figure 25: A region of the fracture surface showing a single repeat of the spectrum.

Plot of the crack growth as measured from the flaw to the outer edge of the cracking.

From the crack growth curve shown in *Figure 25* it can be seen that the five blocks of more severe loading at the start of the test had little impact on the overall crack growth. From the same type of data, and noting the log-linear relationship, extrapolation was used to estimate the life remaining to total failure under residual strength loads.

Whilst numerous cracks existed at the end of cycling, FT46 successfully survived several phases of Residual Strength Testing (RST) [65] including the deliberate severing of two (of the six) vertical tail attachment stubs.

6.1.3 Achievements

Significant technical innovations were achieved during the development and testing phases of the IFOSTP FT46 test, including:

- IFOSTP made first use of test spectra and load sequences derived directly from operational aircraft with a non-linear digital flight control system employing variable control laws and unrestricted manoeuvring;
- development of test loading methods for the stabilators that accounted for large control surface influences in service whilst maintaining a fixed testing deflection;
- the first successful simultaneous application of coordinated dynamic and manoeuvre loads representative of flight conditions on multiple components of a full aircraft test structure;
- development of a unique pneumatic "soft spring" manoeuvre loading system including accurate and rapid response controllers;
- design and development of a controller with 65 actuators of differing type (pneumatic, electromagnetic and hydraulic) with many channels where actuator interaction (dependency) is significant, whilst maintaining strict accuracy levels;
- the development of a novel method to derive loads for residual strength testing [65];
- the quantification of the scatter typical of dynamic buffet loading [66];
- advancement of the field of QF that allowed these techniques, with knowledge of local stresses, to be used to accurately predict component time to failure from limited crack growth information;
- application of evolving technology and use of databases to store, retrieve and catalogue IFOSTP information. This led to several useful fleet management tools; and
- development of possibly the largest archived and documented database of dynamic response data for understanding the aircraft buffet phenomenon and the F/A-18 aft fuselage and empennage structure's response to it.

6.2 The F/A-18 stand-alone FS488 bulkhead tests

The centre fuselage (or centre barrel - CB) of the F/A-18 consists of three fracture critical wing attachment bulkheads. Of these the aft most, known as FS488, was considered the most critical as it suffered early failure in the OEM FSFT known as ST16. Two free-standing (or stand-alone) FS488 bulkhead tests were conducted at DSTO. The testing rig used for this purpose was replicated from that used in similar tests (known as FT40, FT40R, FT41, see [67]) conducted by the OEM. The rig was designed to reproduce the

correct stress distribution in a localised area close to the wing lug attachments. The first DSTO test designated FT488/1 was completed using the OEM spectrum designed for the FSFT of the light-weight centre fuselage (known as ST16). The sequence applied to the bare bulkhead had been truncated to 5,924 load lines, compared to 17,872 strain turning points as recorded by the flange fillet gauge 40165 on ST16. The truncated sequence was scaled to produce what was believed to be the correct peak stress at the critical flange fillet location close to the wing root lug. Several different configurations for this area existed as a consequent of redesign subsequent to the ST16 failure.

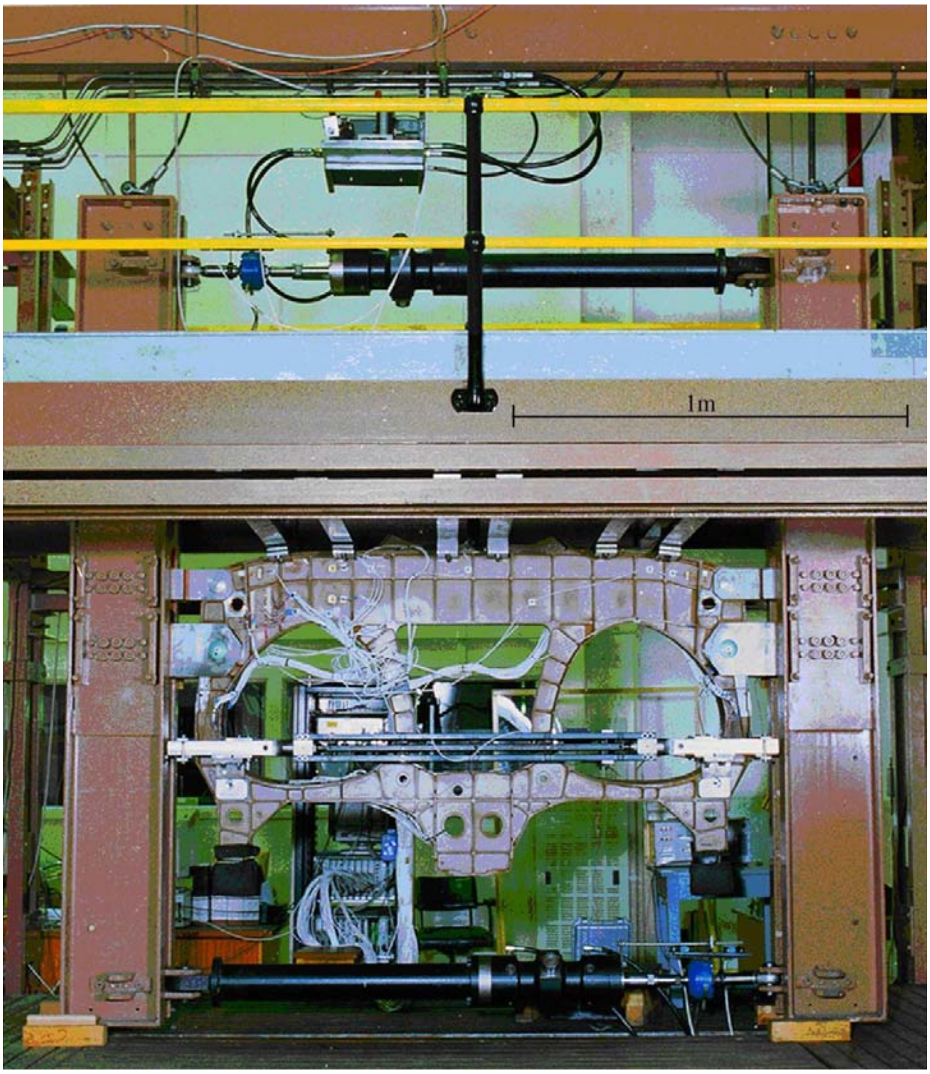


Figure 26: The FT488/2 test rig and test article, looking aft.

The FT40 test, using a bulkhead with the ST16 flange fillet configuration, was to demonstrate that a bare bulkhead test could duplicate the ST16 failure and hence be used to demonstrate the life improvement for other bulkhead configurations. The FT40R test was a repeat ST16 configuration test, and the FT41 test used a six-inch radius flange fillet configuration bulkhead. The DSTO test, FT488/1 completed in 1990 [68], was to determine

the life of an F143 configuration bulkhead reworked to a six-inch configuration bulkhead, which was representative of the rework to thirty-two aircraft in the RAAF fleet.

The second DSTO test, FT488/2 completed in 1999 [69], was a production six-inch configuration bulkhead and used a truncated IFOSTP FT55 (IARPO3a) sequence with the ground loads removed, leaving a sequence of 22,164 lines. The successful outcome of this test was to mitigate against the potential failure of the FS488 bulkhead during the FT55 test and thus compromising the aims of that test. A view of the test rig is shown in Figure 26.

6.3 The P3C aft fuselage test

The P-3 Service Life Assessment Program (P-3 SLAP) was a program of FSFTs and analysis to determine the structural life of the P-3C aircraft (see [70]). The program was a collaborative effort between the USN, RAAF, CF and Royal Netherlands Navy and Air Force. Whilst the major analysis work and wing/fuselage fatigue test was conducted by L-M, on contract to the USN, the other participants also contributed work share. Australia's technical contribution was managed by DSTO and consisted of; a flight loads test program (in conjunction with ARDU and several Australian industry contractors), the teardown of a retired in-service wing (the LH outer wing from the last of the Royal New Zealand Air Force's aircraft updated under their Kestrel program), extensive test interpretation activities, and the conduct of a FSFT of an ex-service P-3 empennage at DSTO. The data obtained from these undertaking were shared with the other SLAP partners and L-M. Other elements of the SLAP program were: an empennage test at L-M of an airframe that had new structure fitted, as part of the Sustained Readiness Program; a landing gear test (sub-contracted by L-M to Vaught Ltd, in Dallas, TX); and wind tunnel testing conducted by the Canadian National Research Council. Post-test teardowns of the main US test article were also carried out by the various SLAP partners. The left hand wing was completed by the Dutch, the right hand wing by the Canadians, the centre wing by the USN and the fuselage and empennage by contractors under the supervision of L-M.

The DSTO empennage test rig development and assembly commenced in 1999 and the rig was commissioned in December 2001 [71]. Test cycling commenced in January 2002 and ran until August 2003. The test article consisted of a retired tailplane, fin and aft fuselage section aft of the aircraft production break at FS1117 (see Figure 27). This was obtained from the USN but had similar hours and date of construction to RAAF aircraft. Rudder and elevators were included to help input loads but were not considered part of the structure under test. The test structure was connected to a length of aft fuselage known as the transition structure which was in turn attached to the test rig in a cantilever arrangement. The test rig was designed and manufactured largely on-site by contractors under DSTO supervision. The control system consisted of the DSTO designed and developed distributed control system, control valves refurbished from the PC-9 test and new actuators built locally by MOOG. The data acquisition system was purchased from CPE Systems under a DSTO specification with the data handling software developed at DSTO. Repairs on the test article were conducted by a mixture of on-site personnel as well as external teams from Australian Aerospace contractors.

The test program consisted of three major parts [72]. The first was the cycling of the test for two lifetimes (total 30,000 hours). The test block applied was a 15,000 hour sequence based on USN 85% severity usage in order to maintain consistency of test spectra across all SLAP tests. Test loads were developed and provided by L-M in the form of jack loads suitable for their empennage test that was being run in parallel. As the DSTO test had been designed ahead of the L-M test, jack locations and numbers (27 in total) were different and so the L-M loads had to firstly be converted to distributions of bending, shear and torque before being re-converted to the required jack loads for the DSTO test.

After reaching the initial test requirement of two lifetimes, little primary structural damage had been found, and a period of augmented test loading was undertaken. The purpose of the augmented testing was to generate structural failures and provide a quick way of inserting additional fatigue damage to cover extended operations of the P-3 aircraft or the need for larger test scatter factors [73]. This consisted of raising all test loads by a factor of 1.6. However, in order to avoid introducing un-representative yielding at known critical locations, the peak loads in the sequence were clipped. This clipping was at a level of 1.1 times the peak load in the baseline sequence for the fin and 1.5 for the tailplane. Such a large load augmentation was both possible and required, due to the large difference between the empennage static limit loads and the test spectrum loads that represented actual in-flight loads experience by the aircraft. One lifetime of this augmented loading was first applied, at the end of which there was a failure of one of the front spar caps at the base of the fin from a fatigue crack that had not previously been detected. Several artificial flaws were inserted into the tailplane at this point to generate additional crack growth data and the test run for a further one lifetime. The third phase of the test consisted of residual strength testing, teardown and inspection. The RST loads consisted of a number of design limit load conditions. A number of additional fatigue cracks were discovered in root locations of the tailplane and fin during teardown. With the fatigue damage left unrepaired, the test article horizontal tail rear spar failed through the inserted artificial flaw at 96.6% of design limit load for the horizontal stabilizer down loading case (which is equivalent to a severe “pull up” manoeuvre in real flight).


Figure 27: Views of the DSTO P3C test

6.4 The F/A-18 centre barrel test

The F/A-18 CB carries wing loads into the fuselage through its three main structural elements, the Y453, Y470.5 and Y488 bulkheads. The three main bulkheads are fracture critical and loss of structural integrity in any of these members may cause the loss of the aircraft. These three bulkheads are made of 7050-T7451 aluminium alloy, which has been coated for corrosion protection with a thin layer of almost pure aluminium by the Ion Vapour Deposition process. As a precursor to this coating, the items to be coated are acid etched to improve coating adhesion. This etching leaves numerous tiny pits in the surface of the coated parts. The IFOSTP FT55 centre fuselage test showed that the safe–life of the CB was insufficient to meet planned withdrawal for all aircraft in the fleet. Thus airframe life recovery activities were required.

A centre barrel replacement (CBR) program was investigated to address some of the deficiencies highlighted by FT55. The USN and CF have already commenced their own CBR programs. For RAAF implementation, two main problems with the CBR program were highlighted; the program may be difficult to run in-country because of logistical concerns, and the availability of aircraft during the program may be insufficient to meet the operational needs of the RAAF. For these reasons, combined with the expected expense of such a program, the RAAF examined alternative strategies to minimise the CBR program.

This strategy, referred to as SRP1++ [74] or Hornet Up Grade (HUG) phase 3, includes a series of modifications made at critical locations late in the life of the F/A-18 to extend the life of some aircraft in the fleet so that they may achieve planned withdrawal date without CBR. The actions that will be taken in SRP1++ were based mainly on experience from the early life of the fleet and the results of the F/A-18 fatigue tests including IFOSTP. Because

of the lack of data from high life fleet aircraft, a number of risks exist in implementing a SRP1++ program, including:

1. Influence of in-service defects including mechanical damage and corrosion.
2. Influence of widespread fatigue damage (and thus potential new fatigue critical locations not previously seen in other F/A-18 fatigue tests).
3. Ineffective repairs.

The teardown and inspection of ex-service CBs was highlighted as a method of reducing the risks involved in a SRP1++ program. Because the USN and CF were then undertaking CBR programs, a number of ex-service CBs were available. Unfortunately the sizes of the average largest cracks present on ex-service CBs are expected to be less than 1 mm [75][76]. This is below the threshold of current practical Non-Destructive Inspection (NDI) methods, meaning that it is difficult to gain service data from ex-service CBs in their current condition. To overcome this obstacle, the Flaw IdeNtification through the Application of Loads (FINAL) program was conducted [75]-[86]. This program involved the application of representative Wing Root Bending Moment (WRBM) fatigue loads to ex-service CBs in a test rig. The fatigue cycling was intended to grow existing flaws to a size where they could be detected under laboratory conditions. After fatigue cycling of each CB had been completed, a teardown including thorough inspection and QF was performed. The data were intended to address the following aims of FINAL:

1. To determine if in-service aircraft contain CB damage not accounted for in the fatigue test program, including mechanical damage and corrosion that are the result of the service environment.
2. To get a more complete picture of the types of defects or degradation that lead to cracking in the fleet.
3. To ensure that future decisions on the CBR program were based on as much relevant information about the structural integrity of the in-service CB as possible; and
4. To provide data to enhance current risk and reliability method deliberations with regard to the F/A-18 aircraft (see [76],[82]).

The test rig, shown in Figure 28, was designed to simulate wing loads at the wing attachment lugs. Pairs of actuators applied equal and opposite loads to the ends of beams that were attached to the sides of each bulkhead. The WRBM produced by the actuators was transferred as a couple at the wing attachment lugs. The CB was rotated by 90 degrees to allow it to sit on one set of beams. The rig was self-reacting so that the top and bottom beams applied equal and opposite bending moments to opposite sides of the test article.

Each actuator was controlled separately, making it possible to proportion the bending moment applied to each bulkhead. This allowed the loading of the CB to be tailored to match in-flight load distributions. It was also possible to continue cycling after individual bulkheads failed by switching off the actuators attached to the failed bulkheads. As a

result, each bulkhead was cycled until failure, maximising the amount of growth to existing flaws.

A truncated version of mini-FALSTAFF (Fighter Aircraft Loading STandard For Fatigue Evaluation) sequence was applied to the test articles. The FALSTAFF loading sequence was developed to represent the standard load history at the wing root of a fighter aircraft [87]. The sequence is equivalent to 200 flights. The normalised sequence was multiplied by the peak IFOSTP FT55 WRBM of 6462 in-kip to produce a WRBM sequence. At the time of writing 3 CBs had been tested.

Figure 28: An F/A-18 centre barrel mounted in the FINAL rig (location of first failure highlighted)

6.5 Testing of the DSTO F-111 Wing Pivot Fitting Optimisation Modification

The F-111 wing pivot fitting is manufactured from a high strength D6ac steel. The pivot fitting (Figure 29) design incorporates stiffening webs and fuel flow vent holes. Some of the stiffener runouts and fuel flow vent holes are prone to fatigue cracking due to residual stresses generated during the application of the CPLT which forms part of the durability and damage tolerance assessment (DADTA) approach used to manage the structural integrity of the aircraft.

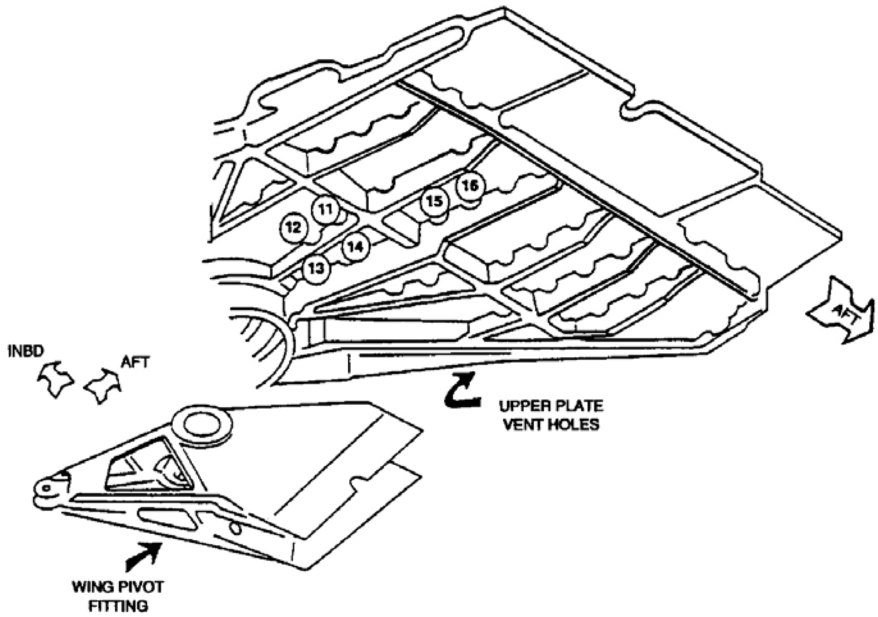


Figure 29: Wing pivot fitting upper plate stiffener runouts and fuel flow vent holes

To alleviate the fatigue cracking problem and associated NDI burden, DSTO designed the wing pivot fitting optimisation modification (WOM). The WOM [88] involved reshaping designated stiffener runouts and fuel flow vent holes to an optimised shape that significantly reduces the level of yielding during CPLT. The WOM was implemented into a limited number of F-111C wings using electro-discharge machining processes and tooling designed by DSTO. This same technique was applied on a surplus F-111A wing (A15-14) obtained from the USAF Aircraft Maintenance and Regeneration Center (AMARC) which was then used in the fatigue rig described in the next section for purposes of validating the life of the WOM configuration. The wing had seen 615 service hours when retrieved from AMARC.

Damage tolerance testing involved progressively introducing more damage to the optimised stiffener runouts and fuel flow vent holes [89]. Initially corrosion pits of a controlled size and depth were introduced at the critical locations followed by the application of blocks of the DADTA2b RAAF DADTA flight mix (see Section 6.6). Unfortunately the wing failed unexpectedly at a known fleet cracking critical location on the outer wing (forward auxiliary spar 281 inches from the pivot fitting – see Figure 31) at a total life of 5,415 hours (only 800 hours from NDI of this area).

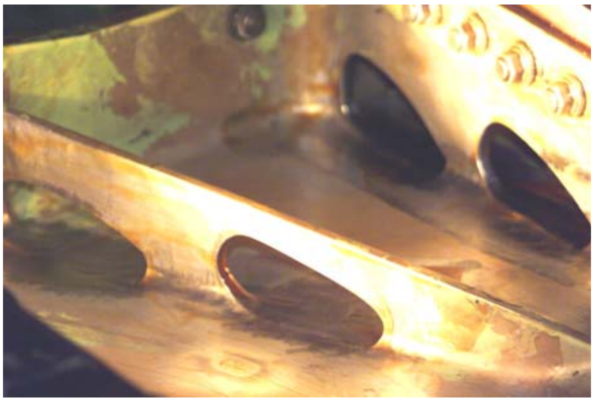
 

Figure 30: Fuel Flow Vent Hole 11, 12, 13 and 14 (left) and Stiffener Runout No 2 optimal shapes

Figure 31: Failure crack of AMARC WOM test

6.6 The F-111 C-Wing Damage Enhancement Test

Whilst the USAF retired its own fleet, the RAAF continues to operate the F-111 aircraft. As part of the RAAF F-111 Sole Operator Program, DSTO was tasked to undertake an ageing aircraft audit of the F-111 aircraft, to establish whether there were any hidden problems that might prevent the aircraft from continuing in service until the planned withdrawal date. Part of this work involved damage enhancement testing of a retired F-111C aircraft wing prior to a teardown inspection to identify all fatigue critical regions of the structure. This program was known as the wing damage enhancement test (WDET) [90]. The retired wing (A15-5) with 25 years of RAAF service was considered likely to have been exposed to environmental degradation that might trigger fatigue cracking.

In addition, undertaking a test of the long wing configuration, applicable to the F-111C aircraft, under loads representative of RAAF service would make up for the lack of such testing by the OEM. The test also provided the opportunity for the performance of DSTO/RAAF designed boron epoxy reinforcements fitted to the wing pivot fitting upper plate and the lower wing skin at the forward auxiliary spar 281 inches from the pivot

fitting (ie the location of failure discussed in Section 6.5) to be evaluated as well as the WOM.

It was deemed necessary to apply loading spanwise and chordwise along the wing to closely represent the bending, shear and torsion load distributions experienced in RAAF service. This required test loads to be diffused into the structure via loading pads and a whiffletree arrangement. There were a number of innovations in the loading arrangement. Firstly, it was the first use in DSTO testing of a two-dimensional whiffle tree (shown in Figure 32), which was common in European fatigue tests. Secondly, single ended hydraulic actuators with an AMRL customised controlled unloading system, were mounted at floor level under the wing, to keep the rig compact and to avoid the practical issues of overhead hydraulic systems. Then, in order to be able to inspect the lower wing skin during the test, the loading pads were fitted so that they could be withdrawn easily. Instead of being bonded to the lower wing skin, they were simply held against the skin by pressure and friction. This was possible because the F-111 has four holes through the wing to attach store pylons. The whiffle tree pads were pressed against the lower wing surface by tensioning a bolt extension of the actuator rod through the pylon mounting hole. When the system was required to apply negative downloading to the wing, a mechanical switch in this pylon mount attachment transferred the load directly from the actuator rod to the wing hard point, bypassing the whiffle tree because it was not bonded and therefore could not apply tension load. The loading system was also required to apply the periodic CPLTs to a limit load condition, which are applied to the RAAF fleet at 2 000 hour intervals. These loads are applied through the wing hard points and so a key was inserted into the whiffle tree attachment switch to effect bypass of the whiffle trees for upload cases as well as download cases in the CPLT cycles.

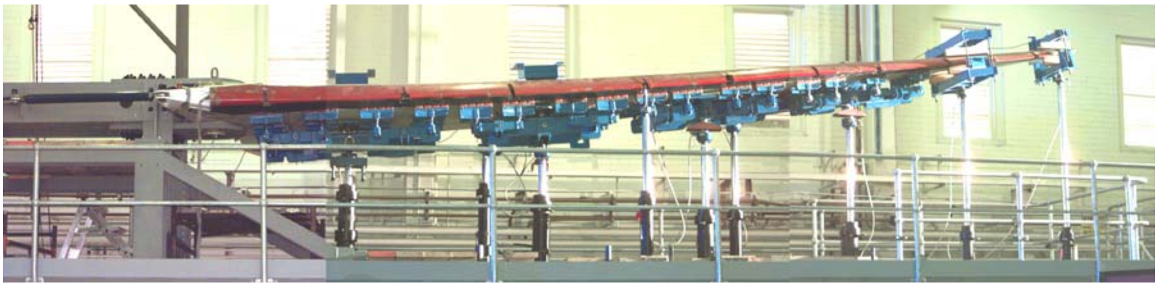


Figure 32: Wing A15-5 in the test rig at 50% of maximum positive proof load

A representative load sequence was derived from flight parameter data recorded using RAAF multi channel recorders. These data were regressed into principal loads using equations developed by the aircraft manufacture. However, the regression load equations were not sufficient to determine the complete spanwise load distributions that were required in order to calculate loading actuator forces. Therefore, a Computational Fluid Dynamic analysis was undertaken at DSTO to supplement the regression load equations to determine full spanwise load distributions. Using the double shear method [92], developed by DSTO for the IFOSTP FT46 fatigue test of the F/A-18 aircraft, actuator forces were calculated.

Due to problems encountered with the original multi channel recorders data sample, some flights were discarded and other flights were manually repaired. This, along with a requirement to ensure that the sample was as representative as possible of flying operations current at the time, led to a re-evaluation of what was then known as the RAAF DADTA flight mix. A new flight mix, known as DADTA2b, was compiled focusing mainly on representing Nz exceedance records, whilst other parameters such as take-off weights, flight duration and mission codes were also considered but not necessarily matched. The DADTA2b spectrum sequence matched typical fleet usage statistics over a 5-year period and represented 500 hours and contained 139,758 loadlines after truncation.

The wing structure failed after only 13,739 hours of combined service and test life, Figure 33. This region of the wing had been inspected approximately 1,500 hours earlier. The failure was unexpected and was attributed to poor build quality associated with a Taperlok fastener hole in the lower wing skin, Figure 34. Teardown and analysis of the wing showed that the build quality problem was widespread and that other holes exhibited crack growth attributable to loading seen whilst the wing was still in service [90]. Based on the test result, a fatigue life analysis showed that many F-111C model wings were already approaching the caclulated interim safe life. The strong likelihood that other wings could contain similar poor build quality led to the decision by the RAAF to source replacement wings for the Australian F-111 aircraft fleet (see Section 6.7).

The A15-5 wing test highlighted the fact that poor surface finish in lower wing skin Taperlok fastener holes, combined with lower than specified levels of interference, can be the precursor to fatigue crack growth effectively reducing the fatigue life of the wing structure. Whilst post-test wing build quality studies may have demonstrated that other wing sets may be of better quality, there was still sufficient evidence of build quality variations and anomalies to warrant the implementation of a safety-by-inspection program, to guarantee the ongoing structural integrity of wing sets. It seemed that problems encountered earlier in the MB326H fleet had manifested here again, only with an increased vengeance. The lower wing skin of the F-111 consisted of 1000's of potential equally critical Taperlok fasteners and an automated NDI system was the only solution.

Figure 33: WDET wing failure at approximately wing station 277

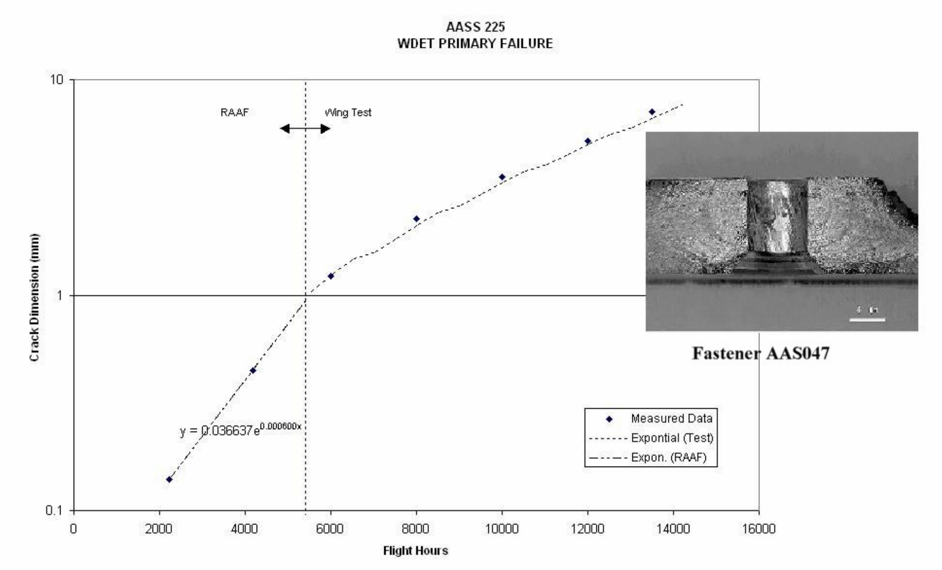


Figure 34: Crack growth from QF of failure site

6.7 The F-111F Wing Economic Life Determination Test

Following the catastrophic failure of the F-111C WDET article (Wing S/N A15-5) in Feb 2002, the RAAF initiated a F-111 wing recovery program. A principal component of this was the procurement from the AMARC and refurbishment of "moth-balled" ex-USAF F-111 D and F model wings. The primary consideration being that these models were produced much later in the production than F-111C wings and were reputedly of better

build quality. However subsequent investigations identified significant deficiencies within the structural certification basis of both the F-111 long wing (i.e. F-111C) and short wing configurations (i.e. F-111 D and F) for the RAAF and USAF operational environments. The F-111 wing economic life (as verified by the manufacturer's A4 fatigue test) was 10,000 hours, however both RAAF operations (long configuration) and USAF operations (short configuration) were shown to have historically been operated significantly more severely than the loading applied to the A4 test. Accordingly that part of the certification basis pertaining to fatigue strength, was no longer considered valid for continuing operation of the RAAF wings up to the planned withdrawal date. As a result of the identified certification basis discrepancies, and significant deviation from the test verification basis evident through assessment of USAF usage data on wings purchased from the USAF, DSTO was tasked by the RAAF (as part of the F-111 Sole Operator Program) to conduct an F-111F Wing Economic Life Determination (F-WELD) fatigue test.

The primary objectives of the F-WELD fatigue test are to determine the economic fatigue life for F-111 D and F model wings as well as providing an opportunity to conduct NDI procedure development activities using an automated ultrasonic inspection system.

The test article was a port F-111F (short) wing, serial number A15-80L, obtained from AMARC after 6,033.9 flying hours. Although A15-80L saw service at several USAF bases, the aim of the test was to provide an economic life and certification basis for F and D wings by replicating Cannon Air Force base usage. Therefore a load sequence was developed that represented typical Cannon short-wing usage and was applied in a rig similar to that described in Section 6.6. This was the first FSFT to be conducted in the new "building 2 structural test facility". At the time of writing, the F-WELD had experienced 18,500 SFH.

6.8 The Hawk Mk 127 test

The Hawk Mk.127 (known as the Lead-In-Fighter, LIF) manufactured by BAe SYSTEMS UK entered RAAF service in 2000. Whilst BAe SYSTEMS had conducted previous FSFTs on T Mk 1 in 1980's, T-45 A Goshawk (USN Carrier variant) in early 1990's and front fuselage of 200 series (single seat) in late 1990's, the results were deemed not to apply to the LIF. In comparison with the T Mk 1, the Hawk Mk.127 is approximately 600 mm longer and 20% heavier. These differences combined with the different RAAF operational flight profiles lead to the determination that a new FSFT was required.

The Commonwealth elected to have the test conducted in Australia by DSTO at Fishermans Bend (see Figure 35) under contract to BAe SYSTEMS with a combined DSTO/BAe SYSTEMS work share and funding arrangement. Personnel from both organisations are participating in the test, along with several Australian contractors with a range of backgrounds and expertise. DSTO analysis was essential to provide specific information on RAAF operational usage of the Australian version of the aircraft for the development of the test spectrum.

Within this agreement BAe SYSTEMS (design authority) maintains responsibility for developing the loads spectra for the FSFT and the design of repairs that will be applied by DSTO. DSTO also has responsibility for the development of the technical systems and conduct of the test. The test rig comprises 1200 data channels, 84 hydraulic actuators and 6 pneumatic volumes (pressurisation of cockpit and fuel tank) with controlled unload on all hydraulic channels using a hydraulic valve pack developed by MOOG Australia based on a previous DSTO design. The electronic control system also incorporates technology developed in the IFOSTP test which was integrated into a commercial system developed by MTS Systems in the USA. Also present is a data acquisition system that can measure 1200 channels, of which 770 are used for various strain gauge elements.

Fatigue testing is due to commence in October 2005 and the test will reach 15,000 equivalent flying hours – which equates to approximately 3,000 actual safe flying hours – before the fleet aircraft reach that point. The full-scale fatigue test is expected to continue until 2012, eight years before the fleet's proposed withdrawal date.

Tailplane (buffet) testing was being conducted separately by BAe SYSTEMS at Brough in the UK at the time of writing using similar technology to that developed through FT46 testing.

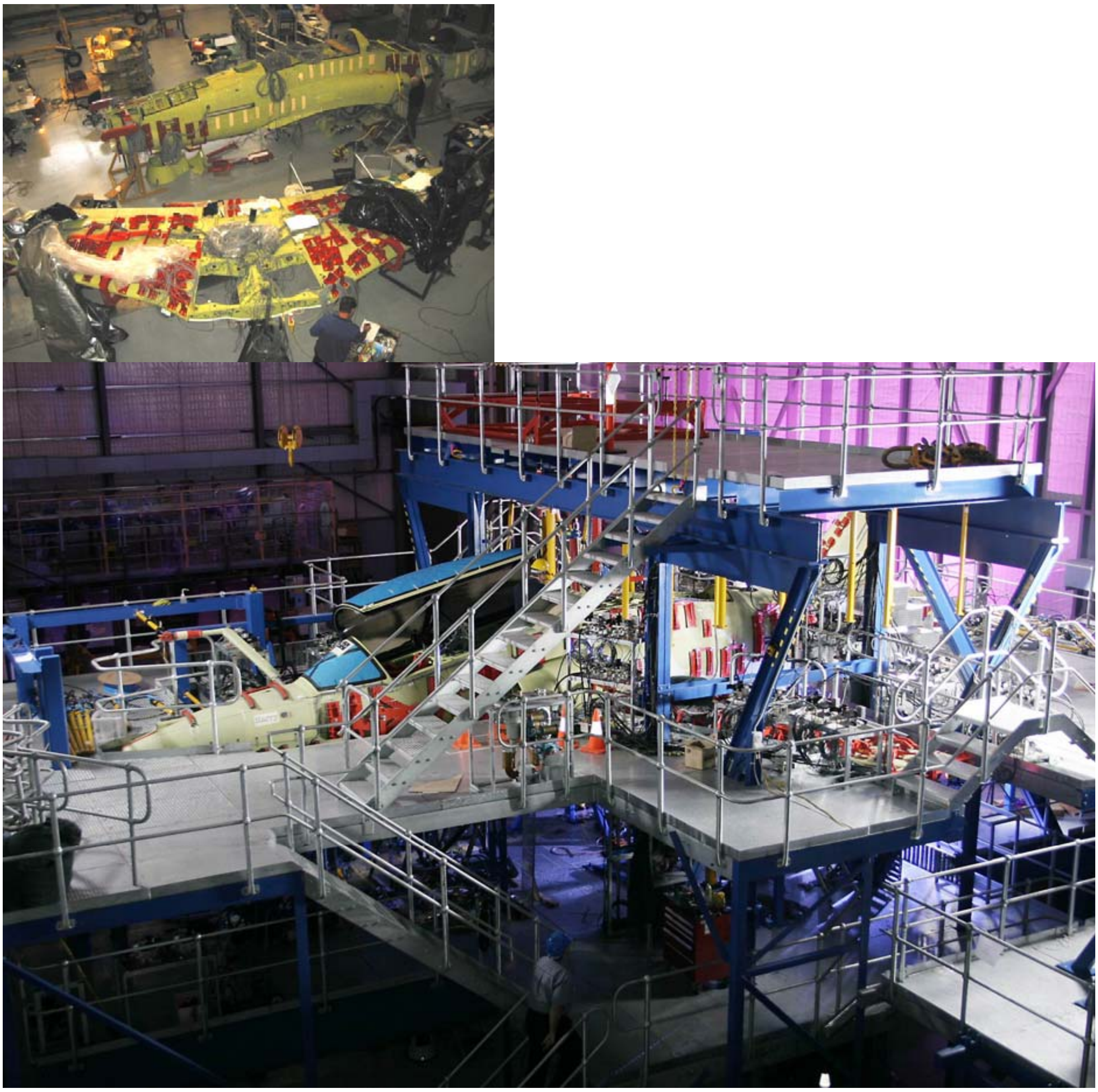


Figure 35: Hawk test preparation (above) and the test rig

7. Conclusion:

The early research at Fishermans Bend on the fatigue of structures and materials won international recognition both as regards its contribution to Airworthiness and to a fundamental understanding of fatigue. These investigations had a significant influence on the development of fatigue design philosophies, theories, testing techniques and technologies, and airworthiness criteria in the aeronautical sphere.

The role played by ARL/AMRL/DSTO in aeronautical fatigue is a very good example of the foresight shown by Wimperis in his report to the Commonwealth Government in 1938,

since the research in this area has enabled the major task of fatigue life substantiation and monitoring to be carried out in Australia with DSTO supplying the R & D input with engineering support from the local aircraft companies.

It will have become apparent that the programme of research on fatigue could only have been achieved by the efforts of a large and very able team of research workers. The assembly and dedication of this team of gifted scientists on such a programme of major importance to aeronautics and to its development in Australia was the direct result of the outstanding qualities of Mr. Coombes as the founder and leader of the laboratories. It was the result of his appreciation of the role that civil and military aviation were to play in the development of the Australian continent; of his ability to win the confidence and collaboration of the civil and military aviation authorities and establishments in the ARL research programmes.

Today's DSTO is also proud of the support given to the through life structural integrity support of (mainly) Australian Defence Force (and to a lesser extent the civil aviation industry) platforms through numerous full-scale fatigue programmes conducted. These have demonstrated that despite better design and analysis tools being available in the latter years, the highly optimised nature of modern aircraft coupled with increased operational requirements will likely lead to surprises, in the way of structural failures, that are better discovered in the test laboratory than through post-accident investigation. The lessons learnt through the Aermacchi in-flight failure investigation, amongst others, should not be forgotten: the next most critical location must always be known.

With the likely prevalence of OEM through-life-support contacts, DSTO's roles in future full-scale fatigue testing is unclear, although its role in the Hawk Mk 127 testing may be one model. Another lesson is also clear, if Australia is to maintain its capacity as a leading airworthiness authority then young practitioners must "cut their teeth" participating in these tests and peering at fracture surfaces under a microscope.

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9. Vale

Dr A.O. Payne sadly passed away in 2004. His legacy however will remain with us for a very long time.

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19. ABSTRACT

This report presents the history of fatigue research at DSTO's Fishermans Bend Australia facility from the early days in the 1940s when Mr. H.A. Wills, Head of the then Structures Division, foresaw with remarkable insight the impending danger of fatigue in aircraft structures. He presented an historic paper at the Second International Aeronautical Conference in 1949 and instituted a comprehensive programme of research on the fatigue of materials and structures which was vindicated within the next decade as fatigue failures began to plague first civil and then military aircraft fleets world wide. DSTO is still a leading world authority on the fatigue of aircraft structures, as many of these research programmes have won international recognition and as fatigue investigations expeditiously undertaken for the RAAF (and at times Civil authorities) have supplied valuable information to the aircraft manufacturers, operators and researchers.